

REQUIREMENT STUDIES FOR THE LAUNCH VEHICLE AND SUPPORT FOR A HORIZON DEFINITION SPACECRAFT

Horizon Definition Study

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SUPPORT FOR A HORIZON DEFINITION SPACE CRAFT

By James O. Barrett
Richard T. LeFrois

HORIZON DEFINITION STUDY

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ABSTRACT

A flight vehicle operations and launch support plan compatible with the recommended spacecraft configuration and flight technique was developed during the Horizon Definition Study. This plan features a two-stage, Improved Delta launch vehicle, Model DSV-3N, and launch operations from the Western Test Range.

FOREWORD

This report documents Phase A, Part II of An Analytical and Conceptual Design Study for an Earth Coverage Infrared Horizon Definition Study performed under National Aeronautics and Space Administration Contract NAS 1-6010 for Langley Research Center.

The Horizon Definition Study was performed in two parts. Part I, which was previously documented, provided for delineation of the experimental data required to define the infrared horizon on a global basis for all temporal and spatial periods. Once defined, the capabilities of a number of flight techniques to collect the experimental data were evaluated. The Part II, documented in this report, provides a measurement program plan which satisfies the data requirements established in the Part I study. Design requirements and the conceptual design for feasibility of the flight payload and associated subsystems to implement the required data collection task are established and documented within this study effort.

Honeywell Inc., Systems and Research Division, performed this study program under the technical direction of Mr. L. G. Larson. The program was conducted from 28 March 1966 to 10 October 1966 (Part I) and from 10 October 1966 to 29 May 1967 (Part II).

Gratitude is extended to NASA Langley Research Center for their technical guidance, under the program technical direction of Messrs. L. S. Keafer and J. A. Dodgen with direct assistance from Messrs. W. C. Dixon, Jr., E. C. Foudriat, H. J. Curfman, Jr., and T. F. Bonner, Jr., as well as the many other people within their organization.

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REQUIREMENT STUDIES FOR THE LAUNCH VEHICLE AND SUPPORT FOR A HORIZON DEFINITION SPACECRAFT

By James O. Barrett
Richard T. LeFrois

SUMMARY

This report describes the results of a study conducted to develop a flight vehicle operations and launch support plan compatible with a flight technique and spacecraft configuration designed for use in a Horizon Definition Study experiment. The goal of this proposed experiment is to make extensive measurements of the earth's infrared radiance profile in the 14- to 16-micron, carbon dioxide absorption band. These measurements will provide the basis for developing improved horizon sensing systems for spacecraft attitude determination.

A previous study defined a basic flight technique and the data requirements necessary to achieve a successful mission. An operational plan compatible with these requirements was developed and features an Improved Delta launch vehicle, model DSV-3N, and launch operations conducted from the Western Test Range. Significant increases in the probability of mission success may be achieved through a multiple-flight concept with scheduled launches of three spacecraft.

INTRODUCTION

Requirement Studies for the Launch Vehicle and Support for a Horizon Definition Spacecraft documented herein are a portion of the Horizon Definition Study (HDS) conducted for NASA Langley Research Center, Contract NAS 1-6010, Part II. The purpose of the Horizon Definition Study is to develop a complete horizon radiance profile measurement program to provide data which can be used to determine the earth's atmospheric state, especially at high altitudes. These data can then be effectively used in many atmospheric sciences studies and in the design of instruments and measurement systems which use the earth's horizon as a reference.

Part I of the HDS resulted in the following significant contributions to the definition of the earth's radiance in the infrared spectrum:

- The accumulation of a significant body of meteorological data covering a major portion of the Northern Hemisphere.
- Computation of a large body of synthesized horizon radiance profiles from actual temperature profiles obtained by rocket soundings.
- Generation of a very accurate analytical model and computer program for converting the temperature profiles to infrared horizon profiles (as a function of altitude).
- An initial definition of the quantity, quality, and sampling methodology required to define the earth's infrared horizon in the CO₂ absorption band for all temporal and spatial conditions.
- An evaluation of the cost and mission success probabilities of a series of flight techniques which could be used to gather the radiance data. A rolling-wheel spacecraft was selected in a nominal 500 km polar orbit.

The Part II study effort was directed toward the development of a conceptually feasible measurement system, which includes a spacecraft to accomplish the measurement program developed in Part I. In the Part II HDS, a number of scientific and engineering disciplines were exercised simultaneously to conceptually design the required system. Accomplishments of Part II of the study are listed below:

- The scientific experimenter refined the sampling methodology used by the measurement system. This portion of the study recommends the accumulation of approximately 380 000 radiance profiles taken with a sampling rate that varies with the spacecraft's latitudinal position.

- A conceptual design was defined for a radiometer capable of resolving the earth's radiance in the 15-micron spectrum, to 0.01 watt/meter²-steradian with an upper level of response of 7.0 watt/meter²-steradian.
- A starmapper and attitude determination technique were defined capable of determining the pointing direction of the spacecraft radiometer to an accuracy of 0.25 km in tangent height at the earth's horizon.

The combination of the radiometer and starmapper instruments is defined as the mission experiment package.

- A solar cell-battery electrical power subsystem conceptual design was defined which is completely compatible with the orbital and experiment constraints. This system is capable of delivering 70 watts of continuous electrical power for one year in the sun-synchronous, 3 o'clock nodal crossing, 500 km orbit.
- A data handling subsystem conceptual design was defined which is capable of processing in digital form all scientific and status data from the spacecraft. This subsystem is completely solid state and is designed to store the 515 455 bits of digital information obtained in one orbit of the earth. This subsystem also includes command verification and execute logic.
- A communications subsystem conceptual design was defined to interface between the data handling system of the spacecraft and the STADAN network. The 136 MHz band is used for primary data transmission and S band is used for the range and range-rate transponder.
- A spacecraft structural concept was evolved to contain, align, and protect the spaceborne subsystems within their prescribed environmental constraints. The spacecraft is compatible with the Thor-Delta launch vehicle.
- An open-loop, ground-commanded attitude control subsystem conceptual design was defined utilizing primarily magnetic torquing which interacts with the earth's field as the force for correcting attitude and spin rates.
- The Thor-Delta booster, which provides low cost and adequate capability, was selected from the 1972 NASA "stable".
- Western Test Range was selected as the launch site due to polar orbit requirements. This site has adequate facilities, except for minor modifications, and is compatible with the polar orbital requirements.

This report contains documentation of the areas of study related to the development of a flight vehicle operations and launch support plan compatible with the mission requirements and spacecraft design concept.

The general outline of the study elements is as follows:

- Selection of launch vehicle and launch site
- Analysis of the effects of using multiple flights on the mission probability of success
- Flight vehicle design restraints
- Flight operations plan
- Launch support system restraints
- Support plan and functional analysis
- Conclusions and recommendations

STUDY REQUIREMENTS AND OBJECTIVES

The following list itemizes the primary and secondary requirements of the Horizon Definition Study.

BASIC REQUIREMENTS

Radiance Profile Measurements

- Spectral interval: $615 \text{ to } 715 \text{ cm}^{-1}$ ($14.0 \text{ to } 16.28\mu$)
- Profile accuracy
 - ▶ Tangent height range: $+80 \text{ km to } -30 \text{ km}$
 - ▶ Instantaneous value of radiance measured must be assignable to a tangent height value to within $\pm 0.25 \text{ km}$.
 - ▶ Radiance characteristics and resolution:
 - Maximum peak radiance = $7.0 \text{ W/m}^2 - \text{sr}$.
 - Minimum peak radiance = $3.0 \text{ W/m}^2 - \text{sr}$.
 - Maximum slope = $0.6 \text{ W/m}^2 - \text{sr} - \text{km}$.
 - Minimum slope = $0.02 \text{ W/m}^2 - \text{sr} - \text{km}$.
 - Maximum slope change = $0.15 \text{ W/m}^2 - \text{sr} - \text{km}^2$.
 - Radiance magnitude resolution = $0.01 \text{ W/m}^2 - \text{sr}$.
 - ▶ Horizontal resolution: 25 km
- Data requirements - Data requirements for the Horizon Definition Study (HDS) experiment, as refined during the study, are as follows:

Minimum requirements. -

- ▶ One-year continuous coverage.
- ▶ "Uniform" time sampling in each space cell over each time cell, i. e., no more than two samples/space cell/day
- ▶ 13 time cells (28 days/cell)
- ▶ 408 space cells

Latitude ($60^\circ \text{ S to } 60^\circ \text{ N}$)	320
Latitude ($60^\circ \text{ N to } 90^\circ \text{ N}$)	44
Latitude ($60^\circ \text{ S to } 90^\circ \text{ S}$)	44

- ▶ Samples per cell

Latitude (0° to 60°)	16
Latitude (60° to 90°)	38
- ▶ Total samples (one year) 110 032

Recommended requirements. -

- ▶ One-year continuous coverage.
- ▶ Maximum of 10° latitude separation between successive samples
- ▶ 13 time cells (28 days/cell)
- ▶ 588 space cells:

Latitude (30° S to 30° N)	128
Latitude (30° N to 60° N)	134
Latitude (60° N to 82.6° N)	96
Latitude (30° S to 60° S)	134
Latitude (60° S to 82.6° S)	96
- ▶ Average number of samples per cell:

Latitude (30° S to 30° N)	45
Latitude (30° N to 60° N)	39
Latitude (60° N to 82.6° N)	67
Latitude (30° S to 60° S)	39
Latitude (60° S to 82.6° S)	67
- ▶ Total samples (one year) 378 508

Mission Profile

Nominal circular, polar orbit of approximately 500 km altitude.

Tracking and Data Acquisition

Limited to the existing Satellite Tracking And Data Acquisition Network (STADAN) with minimum modification.

Experiment Package

- Passive radiometric and attitude measurements with redundancy (more than one unit) in the experiment package for the radiometer and attitude determination device.
- Minimum scan rate >0.5 scans/min average.
- Maximum scan angle with respect to orbit plane $\leq 5^\circ$.

Spacecraft

- Rolling-wheel configuration (spin axis normal to the orbit plane).
- Weight in less than 800 pound class mandatory.

State of the Art

Proven subsystems shall be employed wherever possible.

Mission Effectiveness/Reliability

Reliability shall be approached on the basis of "designing in" successful performance of the one-year, data-collection mission, i.e., the effort is to be biased strongly toward mission effectiveness. Consequently, the mission effectiveness/reliability effort should involve continuing tradeoffs in each sub-function area against the criteria of maximum effectiveness. A numerical estimate of the probable system MTBF shall be made on the final configured system.

Strong consideration should be given to the use of reserve spacecraft as a "backup" means rather than as a continuously ready standby. Specifically, the "backup" concept (as opposed to continuously ready) is of more significance on a Thor-Delta sized vehicle than on a Scout vehicle.

LAUNCH VEHICLE AND SUPPORT REQUIREMENTS

The ultimate objective of this study was to develop a flight vehicle operations and launch support plan compatible with the mission and spacecraft design concept. The preceding lists the initial requirements for this study which, when combined with the following constraints, form the basis for this study.

- Utilize existing launch facilities to the maximum.
- Utilize a booster from the NASA 1972 stable to provide a nominal orbit altitude of 500 km for a spacecraft weight of less than 800 lbs.

- Minimize launch costs in concert with requirement for continuous data coverage over a minimum period of one year.

The study described in this document was conducted concurrently with spacecraft system studies whose purpose was to develop a feasible mission plan and a conceptual spacecraft design. The results of those studies thus formed the ultimate constraints upon the flight operations plan which are discussed in the appropriate following sections of this report.

VEHICLE OPERATIONS

LAUNCH OPERATIONS AND ANALYSIS

Flight Vehicle Studies

The mission performance criteria and the basic system requirements established for the HDS specified nominal spacecraft and orbital characteristics which have, in turn, imposed requirements on the launch vehicle. Considering these requirements, the Scout and Improved Delta families of vehicles were selected as the most desirable choices for the program. These candidate vehicles are discussed in the following paragraphs.

Candidate vehicles. -

Improved delta: The Improved Delta Launch Vehicle is a two- or three-stage, space launch vehicle developed for the National Aeronautics and Space Administration by the Missile and Space Systems Division of the Douglas Aircraft Company. There are currently four Improved Delta configurations: the DSV-3E, which uses a thrust-augmented booster and the X-258 solid motor as the third stage; the DSV-3F, a three-stage vehicle which does not use thrust augmentation; the two-stage DSV-3G, which uses a thrust-augmented booster; and a two-stage DSV-3H vehicle, which does not incorporate thrust augmentation. For this study, a fifth configuration soon to be operational was also considered. It is the DSV-3N, which is a two-stage vehicle with larger first-stage tankage and a thrust-augmented booster.

Scout: The Scout Launch Vehicle is a four-stage, space, launch vehicle developed for NASA by the Astronautics Division of the LTV Aerospace Corporation. The current Scout configuration consists of four solid-fuel stages with the FW-4S as the standard fourth stage.

A feasibility study is currently being conducted by LTV under NASA contract to evaluate an improved first-stage version of the Scout vehicle. This version provides an increased payload/orbit altitude capability.

Launch site selection. - Facilities to accommodate Delta launches are available at both the Eastern and Western Test Ranges (ETR and WTR). Scout launches may be conducted from either the WTR or the NASA Wallops Island Facility. However, specification of a near polar (inclination 97.38°) orbit for the HDS mission produces a clear advantage for WTR as the launch site. A launch southward into a retrograde orbit from either the ETR or Wallops Island would suffer severe range safety constraints with corresponding reductions in payload capability.

Launch vehicle selection. - The HDS mission concept imposes a requirement for a sun-synchronous circular orbit at an altitude of 500 kilometers (270 n. mi.). The basic system requirements specify a payload weight not in excess of 800 pounds. These requirements are the primary factors influencing the selection of the launch vehicle.

Figure 1 illustrates the capabilities of the Delta and Scout launch vehicles in terms of the payload which can be injected into a circular polar orbit from the Western Test Range. The current Scout vehicle designed for the specified mission altitude is limited to payloads of less than 300 pounds. Since this weight limitation is not compatible with the currently defined system requirements, the Scout vehicle cannot be considered as a candidate launch vehicle within these constraints. However, during the progress of this study, some distinct operational advantages to using the Scout vehicle became apparent, and these are discussed in a subsequent section of this report.

Several Delta launch vehicles are compatible with the desired 800-pound payload capability, but the orbit-injection, accuracy requirements appear most compatible with the direct-injection, ascent mode. This added requirement narrows the choice to either the 3-stage Delta (DSV-3E) or the 2-stage Delta (DSV-3N).

Further consideration of the orbit-injection accuracy requirements reveals the following advantages of the 2-stage over the 3-stage vehicle:

- Improved mission accuracy - The two-stage deviation in inclination and apogee to perigee distance is estimated at 0.06 degrees and 11 n. mi. (200 n. mi. circular orbit), respectively. This represents a nominal 300 percent improvement in inclination deviation and 100 percent improvement in apogee to perigee deviation compared to a three-stage version.
- Reduced spacecraft costs because of:
 - (a) Reduced spin and static/dynamic balance requirements for spacecraft.
 - (b) Lower acceleration and vibration environment for system and component design.
- Increased fairing volume envelope. With the elimination of the third stage, the inside envelope is increased approximately 15 percent.
- Increased mission reliability through simplicity.

In view of the above considerations, the 2-stage Delta vehicle, DSV-3N, has been chosen for the HDS launch vehicle. This vehicle is capable of orbiting an HDS spacecraft weighing up to 800 pounds by a direct-injection, ascent mode. The payload capability can be increased to 1200 pounds by using a Hohman transfer.

The DSV-3N vehicle is scheduled to be operational by late 1968 and will be available for the HDS mission in the planned 1971-72 time period.

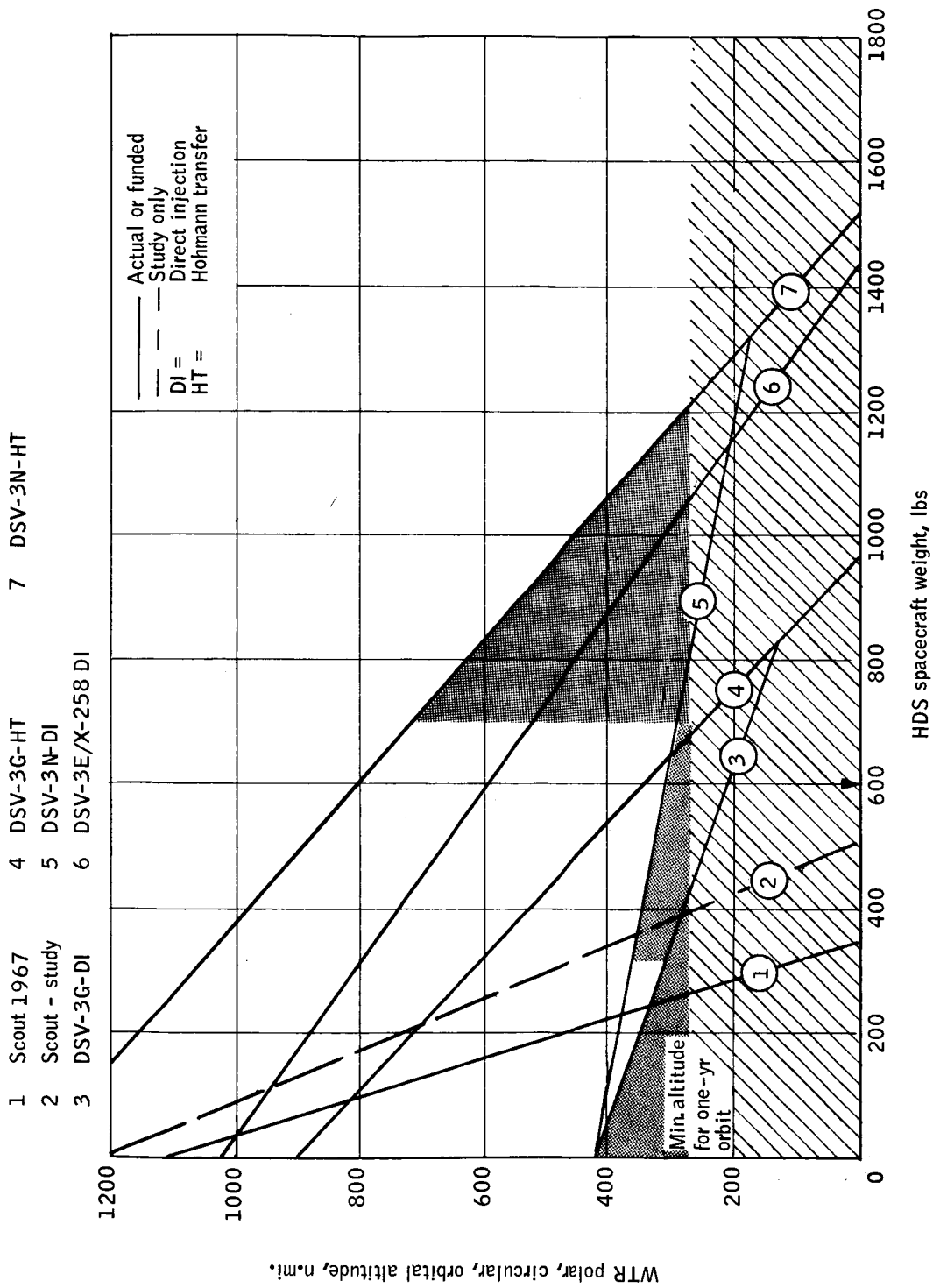


Figure 1. HDS Spacecraft/Booster Vehicle Performance Capabilities

LAUNCH OPERATION STUDIES

A consideration which must be evaluated in the choice of a launch vehicle is its compatibility with the mission operational constraints. The HDS mission objectives include the requirement to sample radiance profiles over a continuous, one-year period. To determine a probability of success in attaining these objectives, a study was conducted of an operational concept involving reserve spacecraft to be launched in the event of failure of the orbiting spacecraft. For the purpose of this discussion, reserve spacecraft are defined as flightworthy units which will be maintained at a prescribed state of "launch readiness" for possible launch on an "as needed" basis.

A fundamental constraint is imposed upon this concept because only one launch pad is available to support Delta vehicle launches from the WTR. Thus, reserve spacecraft launches of this program must compete with Delta launches of other programs, and a spacecraft and launch vehicle cannot be maintained in a continuous "pad-ready" status.

Two significant study elements were:

- The number of reserve spacecraft required to provide a given probability of mission success.
- The required state of launch readiness.

Number of Reserve Units Required

The number of reserve spacecraft and launch vehicles required to insure a year of operation is highly dependent upon the anticipated spacecraft failure rate and the launch vehicle reliability. This requirement was evaluated for the HDS mission through a statistical probability analysis aided by a technique termed State Diagramming and a contractor developed computer program, State Interpretation Program (SIP). A detailed description of the State Diagramming technique is presented in Appendix A.

Briefly, State Diagramming is a generalized reliability modeling technique which facilitates programming such problems on a computer. A typical state diagram is shown in Figure 2, depicting a situation having a spacecraft in orbit and three ready reserve vehicles.

In the example, any of five states or conditions are possible. State E1 represents the initial condition, and each succeeding state represents the condition brought about by failure of the operational spacecraft. The functions shown in the diagram do not represent the mathematical operations to be performed; they represent the inputs to the SIP computer program by which the problem identified in the diagram is solved. The direct paths from each state to the succeeding state are shown to be dependent on both the spacecraft failure rate and launch vehicle reliability, with the exception of the last path which depends upon spacecraft failure rate only. Indirect paths from one state to another are dependent upon the additional factor of the probability of launch vehicle failure.

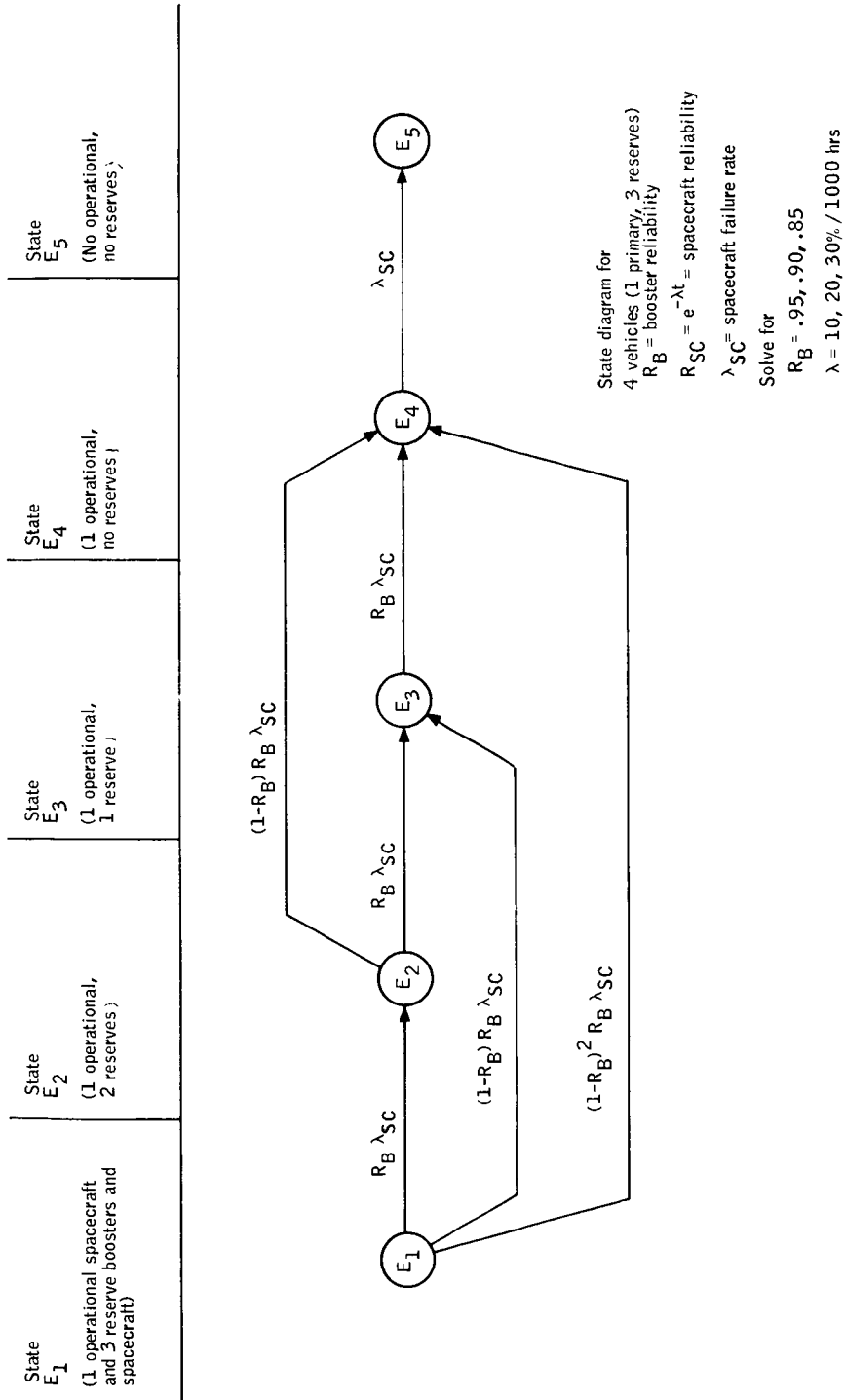


Figure 2. Typical State Diagram

Once the state diagram illustrating the possible conditions was drawn, the problem was programmed using the SIP to obtain parametric results of three different spacecraft failure rates and launch vehicle reliabilities for various numbers of space vehicles. A typical set of results is shown in Table 1 in which the launch vehicle reliability was assumed to be 90 percent, the spacecraft reliability was based on a failure rate of 10 percent/1000 hours, and three spares were programmed.

The columns in the table contain the following information:

- TIME - The time is in hours and is given for $T = 0$ and at the end of each month for twelve months.
- REL - 1 SC - This is the classical reliability at the end of each time period. It represents the probability of one spacecraft (of the four possible) being operational at that time. The example shows a 97 percent chance that a spacecraft will be operational at the end of one year, provided three reserve units are made available.
- FOM - 1 SC - This is the figure-of-merit (FOM) reliability, i.e., the percentage of data that will probably be collected from the total amount of data the spacecraft is capable of collecting if no failures occurred. These numbers are more appropriate than classical reliability for the HDS mission since data collection is more important than having the spacecraft operational at the end of one year.
- STATE - 1, 2, 3, 4, 5 - These represent the probabilities of being in a particular state. The sum of all states is one since spacecraft must be in one of the five states at all times.

Representative examples of the parametric results are summarized in Figures 3 through 7. Additional figures containing information supplementary to this discussion are contained in Appendix B.

Figure 3 is a plot of the classical reliability function in a program for which no spares are provided. These curves represent the probability that the spacecraft will be operational at any given time in a one-year period. The significance of failure rate is evident.

Figures 4 and 5 are presented to show the effect of spacecraft and booster reliability, respectively, on the number of reserve spacecraft provided. In Figure 4, the curves show mission FOM reliability for two spacecraft failure rates and a booster reliability of 90 percent. As seen in the figure, a program in which three reserve vehicles are planned, coupled with a spacecraft failure rate between 10 and 30 percent per thousand hours, provides a better than 90 percent probability of obtaining the mission data requirements.

Figure 5 shows that the difference between a booster with a reliability of 0.85 and one of 0.95 does not appreciably affect the mission success for a given spacecraft failure rate.

TABLE 1.- TYPICAL STATE INTERPRETATION PROGRAM RESULTS

Booster reliability = 0.90

Spacecraft failure rate = 10%/1000 hrs

TIME	REL-ISC	FCM-ISC	STATE-1	STATE-2	STATE-3	STATE-4	STATE-5
.00	.9999C00CC	1.000000000	.900000000	.090000000	.009000000	.000900000	.000100000
73C.00	.999760727	.999842608	.836701824	.138714544	.021274164	.003070196	.000239273
146C.00	.999429136	.999727058	.777855492	.180165220	.035239679	.006168745	.000570864
219C.00	.998837869	.999535989	.723147898	.215130472	.050379818	.010179681	.001162131
292C.00	.997921672	.999252792	.672287962	.244315340	.066255895	.015062476	.002078328
365C.00	.996618961	.998861441	.625005071	.268357923	.082497982	.020757985	.003381039
438C.00	.994872993	.998346729	.581047648	.287835298	.098796602	.027193445	.005127007
511C.00	.992632707	.997694433	.540181808	.303268936	.114895294	.034286670	.007367293
584C.00	.989853276	.996891450	.502190115	.315129662	.130583977	.041949522	.010146724
657C.00	.986456435	.995925887	.466870428	.323842189	.145693030	.050090788	.013503565
730C.00	.982530618	.994787122	.434034821	.329789267	.160088017	.058618512	.017469382
803C.00	.977930940	.993465834	.403508585	.333315481	.173664991	.067441884	.022069059
876C.00	.972679058	.991954011	.375129299	.334730714	.186346319	.076472726	.027320942

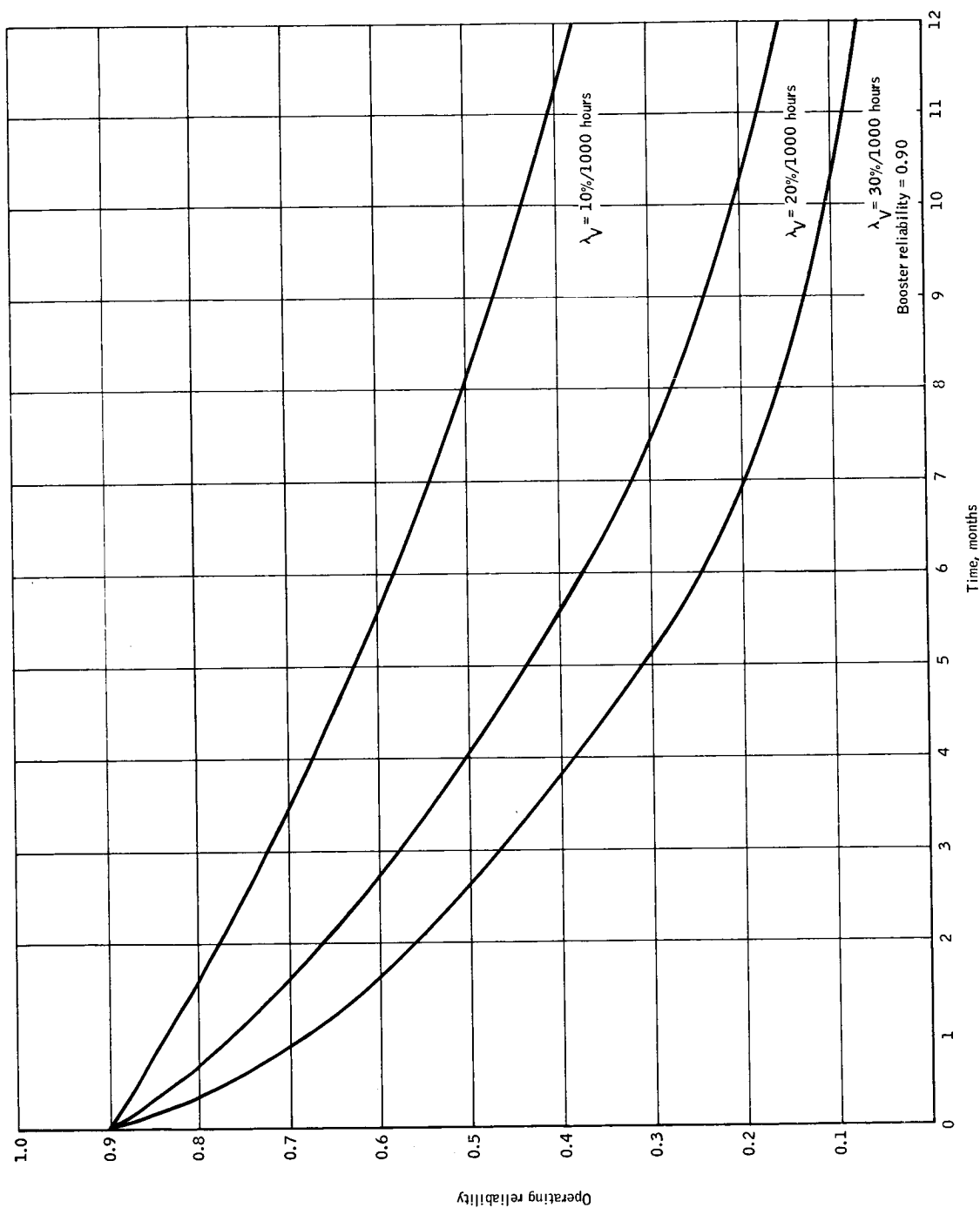


Figure 3. Effect of Spacecraft Reliability on Mission Success
With No Reserve Units

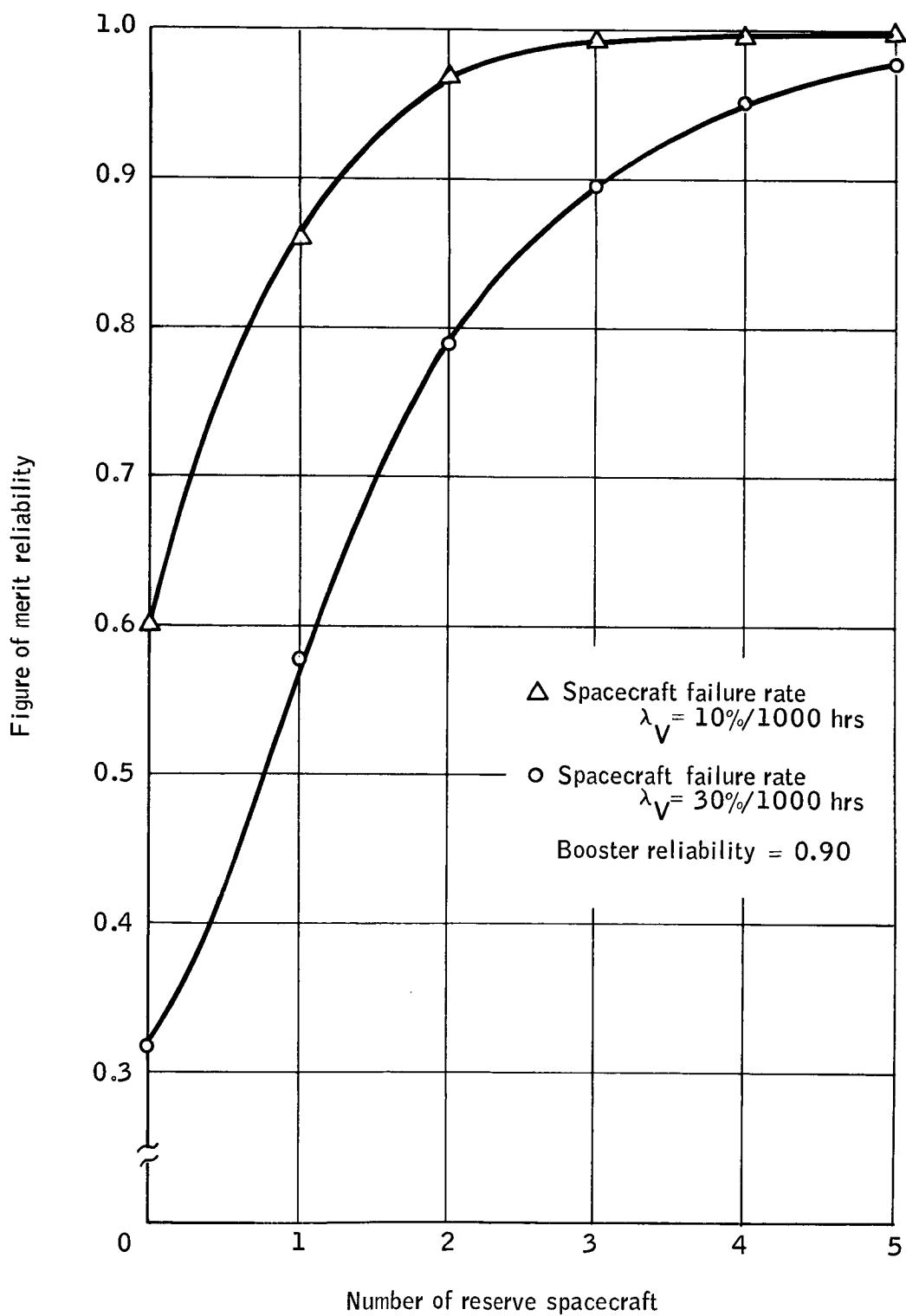


Figure 4. Effect of Spacecraft Failure Rate on Number of Reserve Spacecraft

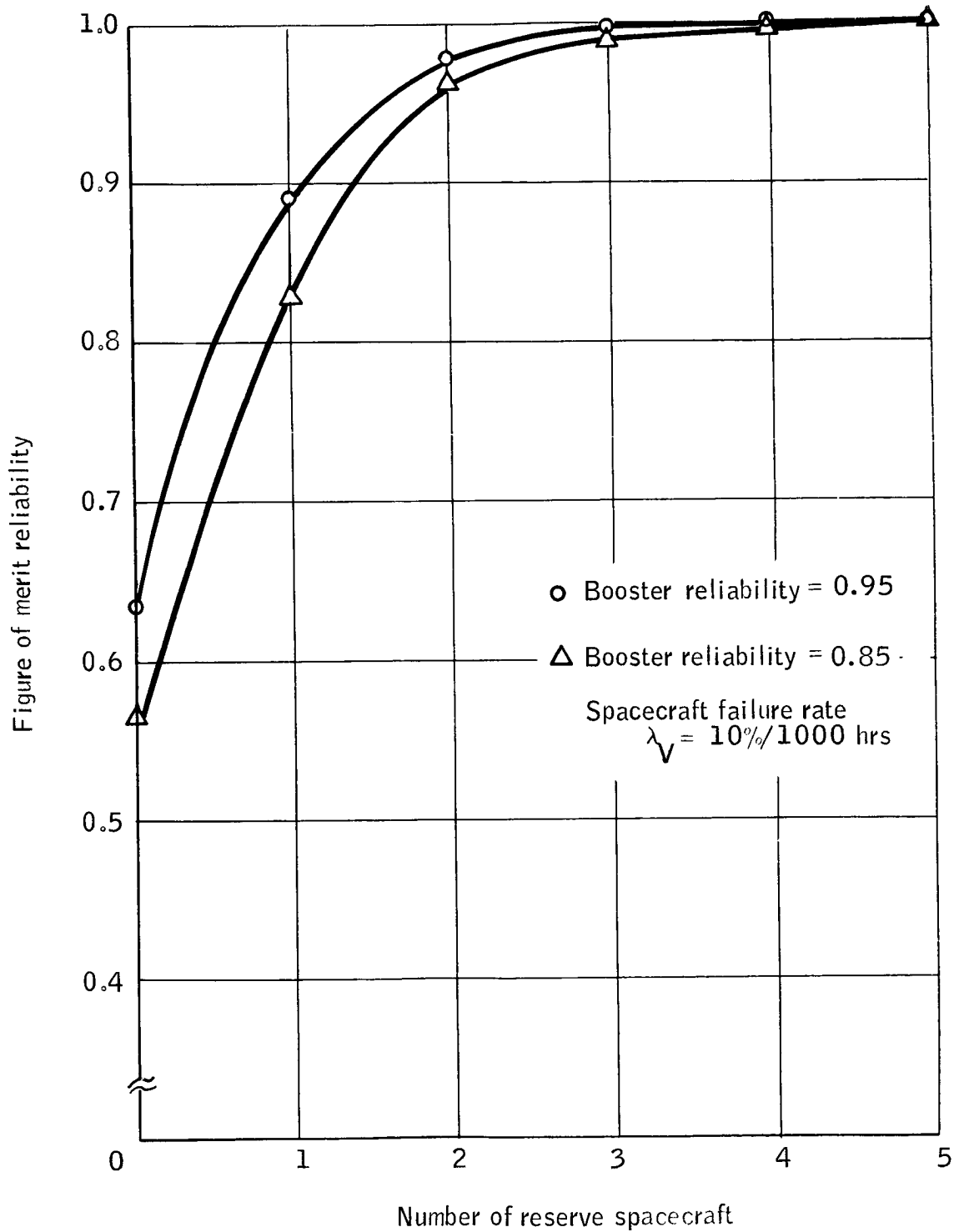


Figure 5. Effect of Booster Reliability on Number of Reserve Spacecraft

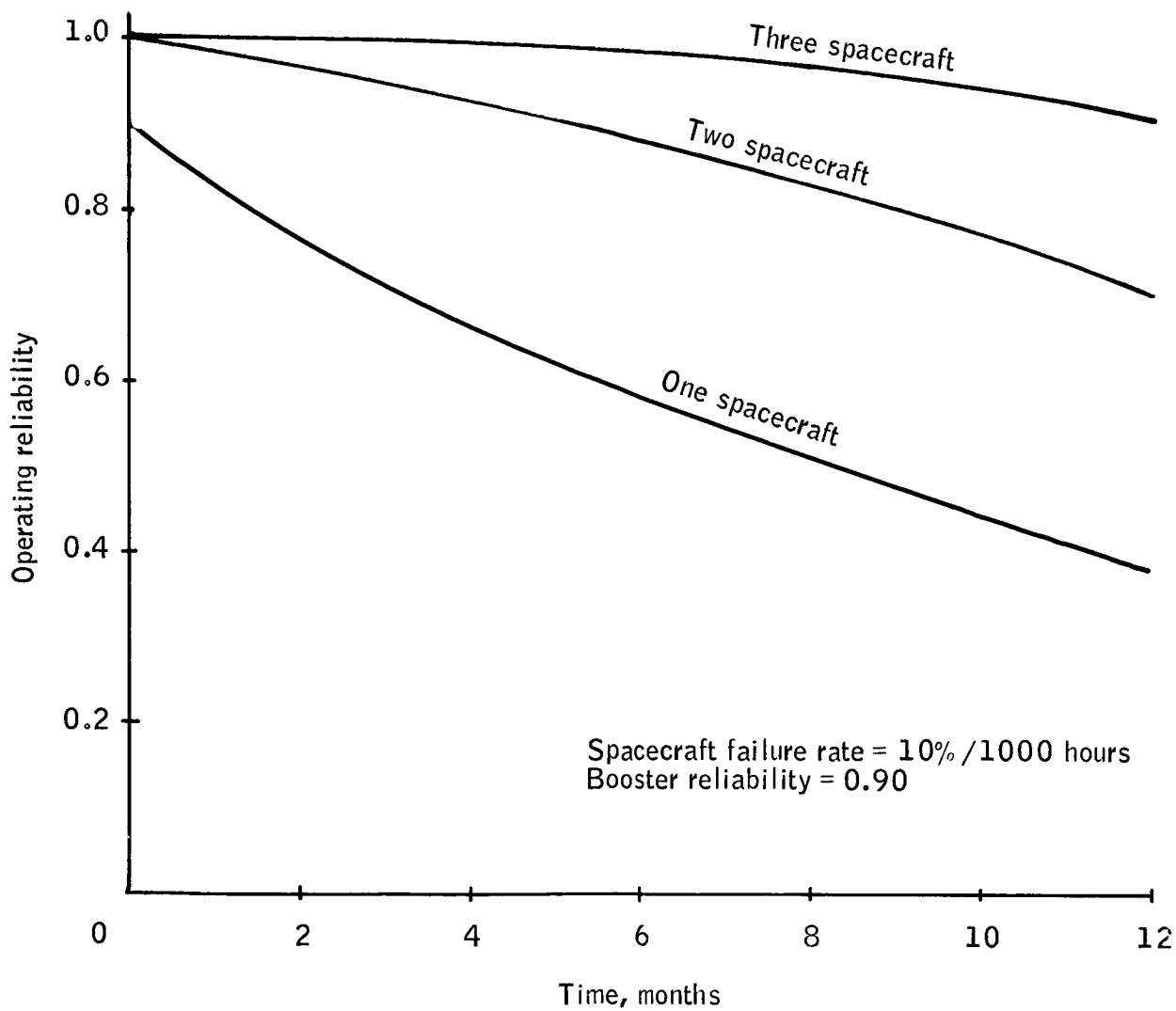


Figure 6. Mission Reliability as a Function of Operating Time

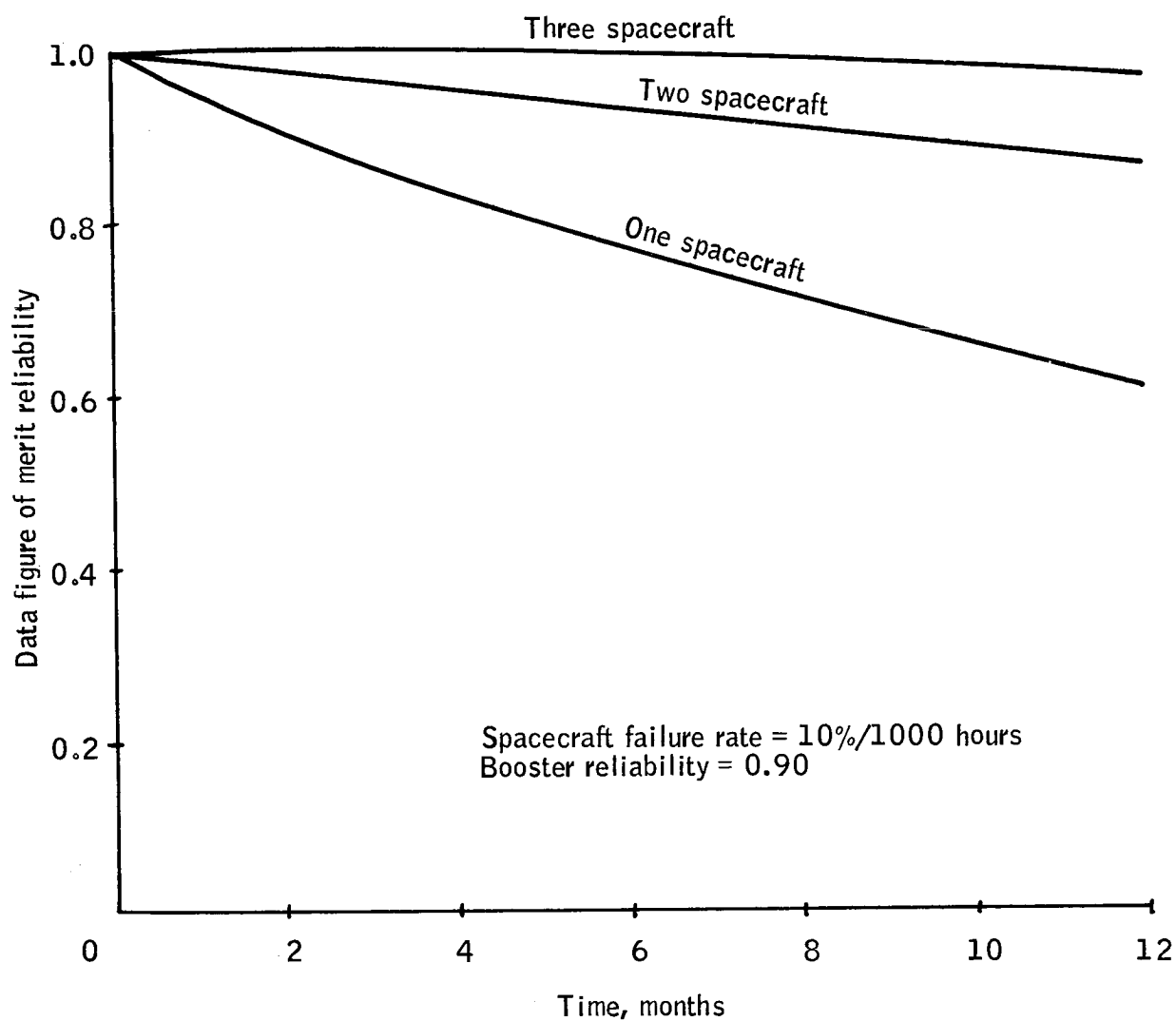


Figure 7. Figure of Merit Reliability as a Function of Operating Time

The significant effect of reserve units on the mission reliability and data FOM as a function of the operational time period is illustrated in Figures 6 and 7.

Figure 6 presents curves of the classical reliability function representing the probability that a spacecraft will be operational at any given time in the year, assuming various numbers of planned reserve units. The curves are drawn for an assumed spacecraft failure rate of 10 percent/1000 hours and a booster reliability of 0.90. Figure 7 presents a similar curve for the mission FOM reliability.

These curves show that for the assumed spacecraft failure rate the probability of a spacecraft successfully completing the one-year mission is only approximately 38 percent with a single planned spacecraft, but increases to 63 percent with one spare and to 82 percent with two reserve units. Similarly, the probabilities of obtaining the data gathering objectives are approximately 60 percent with a single spacecraft, 86 percent with one reserve, and 98 percent with two reserve spacecraft.

The significant conclusions which can be drawn from these curves are:

- The reserve spacecraft concept quickly approaches a point of diminishing returns.
- The lower the spacecraft reliability, the more reserves will be needed to accomplish a given mission success.

The figures presented provide a means for selecting the number of reserve spacecraft on the basis of mission reliability requirements. However, the utility of a reserve concept is also dependent upon the ability to launch these reserve spacecraft within a reasonable period of time following failure of the operational spacecraft, a factor termed launch readiness.

Launch Readiness

The importance of launch readiness time to the HDS mission is dependent upon the maximum amount of interrupted or lost data which can be tolerated. This is a factor which cannot be specified definitively since it is primarily data dependent, i. e., it depends upon the unknown degree of correlation between adjacent data samples and the time of year in which the data interruption occurs.

The constraint imposed upon the state of launch readiness results primarily from launch vehicle considerations which differ significantly between the Delta and Scout vehicles.

The Delta launch vehicle is assembled and checked at the Douglas Aircraft Company facility in Culver City, California and trucked to the Western Test Range. Currently, the prelaunch operations occupy approximately 60 to 90 days.

The Delta vehicle office of the NASA GSFC estimates that in the 1971-72 time period of the HDS program, the corresponding vehicle preparation time will be reduced to 30 days. No capability for allowing the booster to be maintained in a more ready status is planned, and a requirement to reduce the launch readiness time below 30 days will incur significant additional program costs.

Facilities at the WTR to support Scout launches allow this vehicle to be maintained in a 7- to 10-day state of readiness for no additional support costs.

Operational Cost Comparisons

In order to evaluate realistically an operational concept from a total program viewpoint, some method of cost comparison must be employed. The primary factors involved in developing such a concept have been shown to be:

- Spacecraft failure rate,
- Booster reliability, and
- Launch readiness.

Spacecraft and launch operation costs are related through these factors to the probability of mission success. Study results also provide evidence that a tradeoff exists between the cost to provide a high level of spacecraft reliability, the cost to provide spare spacecraft, and the probability of having to use these spares to successfully fulfill the one-year data-gathering requirement. A normalized cost comparison was conducted to illustrate the relationship between mission success and the operational and economic elements of the HDS program. This comparison is purely illustrative and is in no way intended to provide a means of cost estimation.

Figure 8 presents estimated unit Delta vehicle launch costs as a function of the number of planned launches in a one-year period. The figures shown were normalized to the single-unit costs which are based on a projection of current Delta costs to the 1971-72 time period as provided by the Delta vehicle office of the NASA GSFC. The unit costs decrease with an increase in planned launches since the launch service costs are prorated over the operational year on the basis of the number of vehicles launched. The effect of this factor was accounted for by reducing the single-unit costs along a 96-percent learning curve.

Figure 9 presents estimated unit spacecraft hardware costs as a function of design life. The cost for a single unit designed with a one-year life was selected as unity. An 85-percent learning curve was used to estimate costs for additional one-year lifetime units. To illustrate the effect of "designed in" reliability on the spacecraft costs, it was assumed that system redundancy in a 12-month spacecraft represented 40 percent of its costs. Therefore, the unit cost of a spacecraft with minimum redundancy is 60 percent of the cost of a one-year spacecraft.

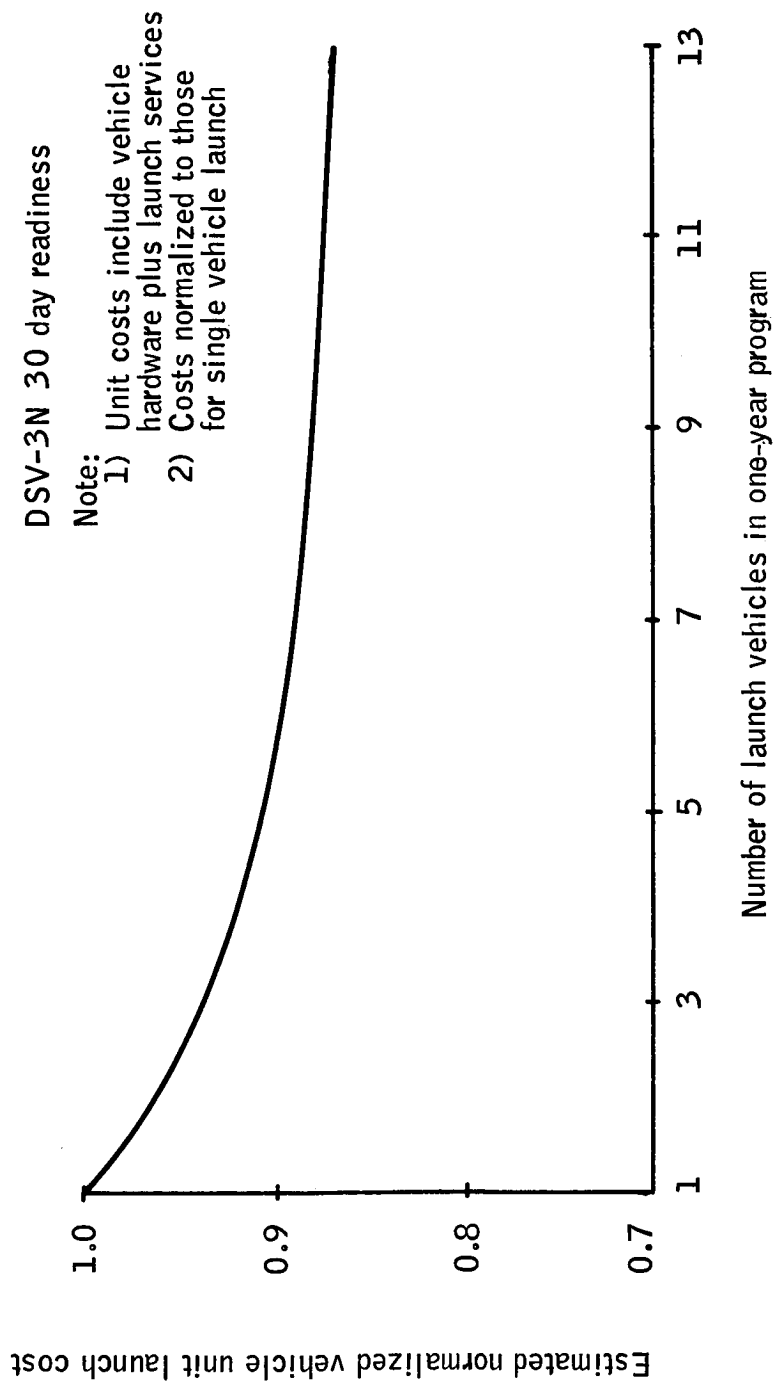


Figure 8. Estimated Delta Vehicle Launch Costs

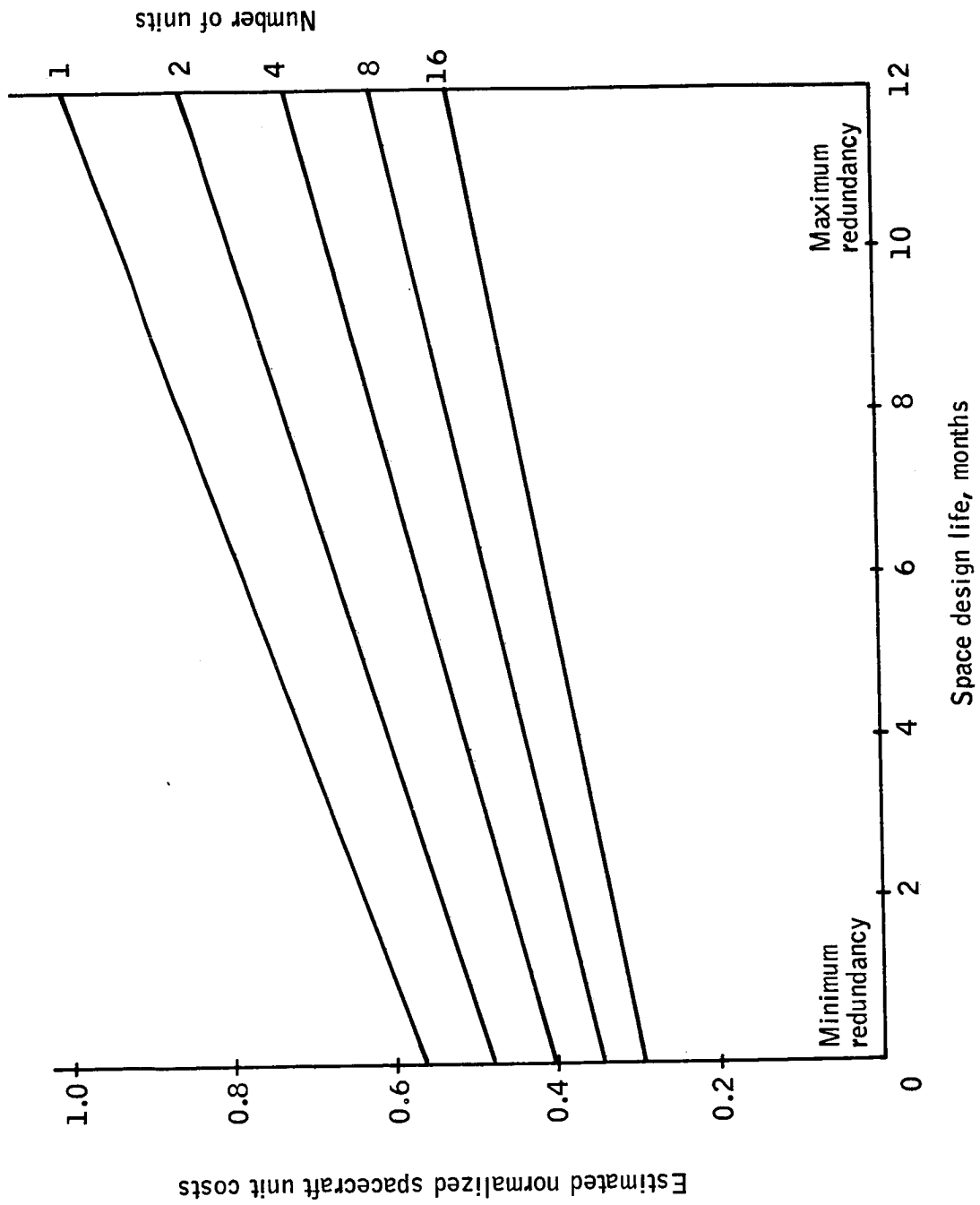


Figure 9. Estimated Spacecraft Cost Profile

Figure 10 illustrates the estimated variation in launch operation costs associated with minimum, nominal, and maximum launch vehicle readiness. The nominal launch readiness condition of the Delta vehicle, as discussed previously, is 30 days. The vehicle would be stored in an assembled and tested condition pending delivery to the launch site. The maximum launch readiness condition is one requiring maintenance of the booster on the launch pad in a tested and ready status. The requirement for a full time maintenance crew would incur a significant cost increase over the nominal case. The minimum readiness condition of greater than 90 days would be provided through normal call-up and would affect some cost reduction over the nominal, since pre-assembly, checkout, and storage are not involved.

Cost model. - The cost of launch operations for a program involving a given number N of launch vehicles may be shown as:

$$CLO_N = N (UC_V)_N (LRF) + N (UC_S)_N \quad (1)$$

where:

CLO_N = Cost of launch operations in a program involving N launch vehicles and spacecraft

$(UC_V)_N$ = Unit cost of N th launch vehicle

$(UC_S)_N$ = Unit cost of N th spacecraft

LRF = Launch readiness cost factor

From the above cost model and the reliability figures previously discussed, Figure 11 was prepared to show the relationship between operational costs and mission performance. It should be noted that the cost model does not reflect total program costs but rather depicts only the incremental costs involved in the launch operations and the effects on those incremental costs of variations in the launch operations approach. The significance of the number of reserves provided, spacecraft failure rate, and launch readiness on mission success and costs is evident.

The figure shows for example that for nominal conditions of 30-day readiness, spacecraft failure rate of 20 percent/1000 hours, and booster reliability of 0.90, an approximate 86-percent probability of mission success may be achieved by providing three reserve spacecraft. Furthermore, the operational costs with three reserves will be approximately 3.3 times that for no reserves. If the spacecraft failure rate were reduced to 10 percent/1000 hours, then, for the same costs, the probability of mission success would improve to approximately 97 percent.

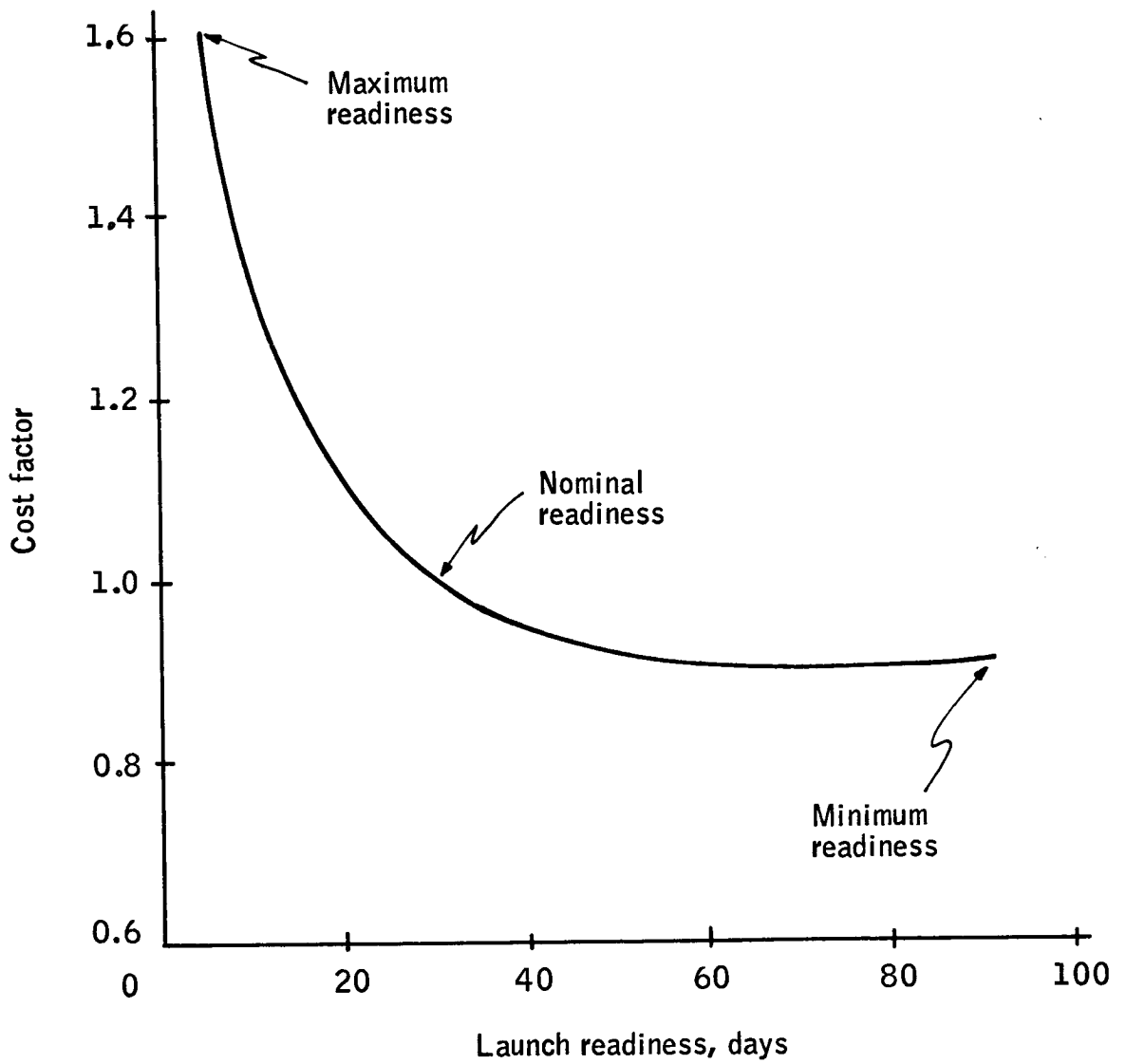


Figure 10. Estimated Delta Vehicle Launch Readiness Cost

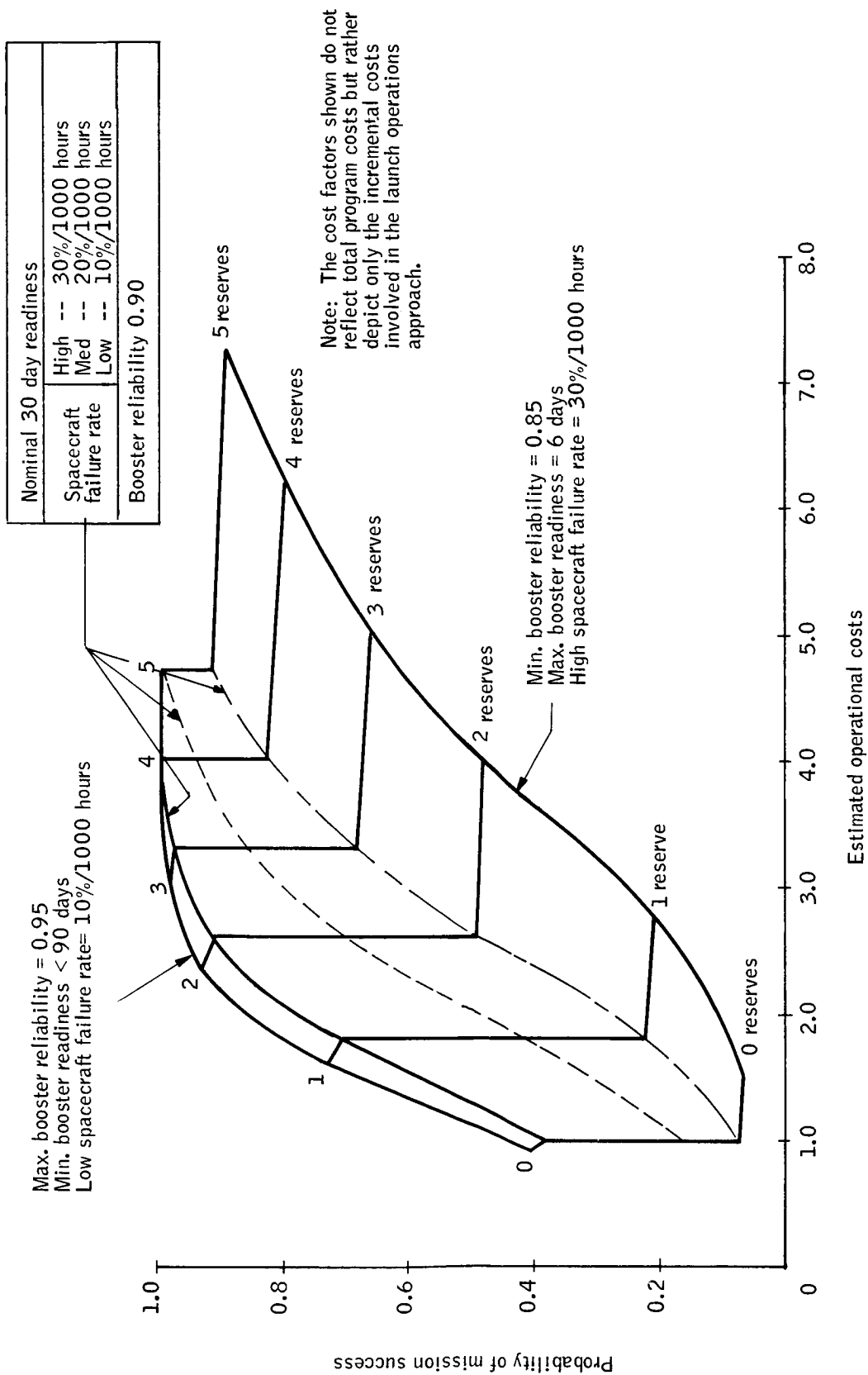


Figure 11. Estimated HDS Operational Cost versus Performance Envelope

The study results provide a means for determining the number of reserve spacecraft on the basis of the required probability of mission success. It is apparent that the spacecraft reliability is a primary factor in achieving success and must be made as high as practical by care in the design, construction, and testing of the spacecraft. However, since there are practical limits on the achievable spacecraft reliability, reserve spacecraft are necessary. The operational method by which those spacecraft might be employed is of interest.

Operational concepts. - The operational concept assumes that random failures determine the probability that the spacecraft will successfully complete the one year mission. The spacecraft is therefore as likely to fail at one time as at any other, and the probability of success may be described by an exponential function as:

$$P_s = e^{-\lambda t} \quad (2)$$

where t is the operating time and λ is the spacecraft failure rate. Since the spacecraft must be launched into orbit, the reliability of the booster is introduced, and the total probability is the product of the two factors.

Figure 12 shows the probability of success as a function of operating time for a spacecraft with an assumed failure rate of 10 percent per thousand hours and a booster reliability of 95 percent. The figure reiterates what was revealed in the launch operation studies, namely that the probability of survival of a single spacecraft for a year is so poor ($P_s < 40$ percent) that major system redundancy is required.

Within the spacecraft design constraints, the addition of major subsystem redundancy is impractical. Therefore, the probability of success must be improved by employing redundant (standby) spacecraft.

Simultaneous Launch of Multiple Spacecraft. - One operational approach might be to launch simultaneously more than one spacecraft with a single booster. The probability of survival of N simultaneously launched spacecraft can be computed as

$$P_{s_N}(t) = 1 - \left[P_{f_1}(t) \right]^N \quad (3)$$

where $P_{f_1} = 1 - P_{s_1}$ = probability of failure in a single spacecraft.

Curves of this function for two and three spacecraft launched simultaneously were added to Figure 12. It may be seen that simultaneous launch of two spacecraft increases the probability from 40 percent to 62 percent that at least one will survive to the end of a year. Similarly, three simultaneously launched spacecraft show a 76-percent probability that at least one of them will survive.

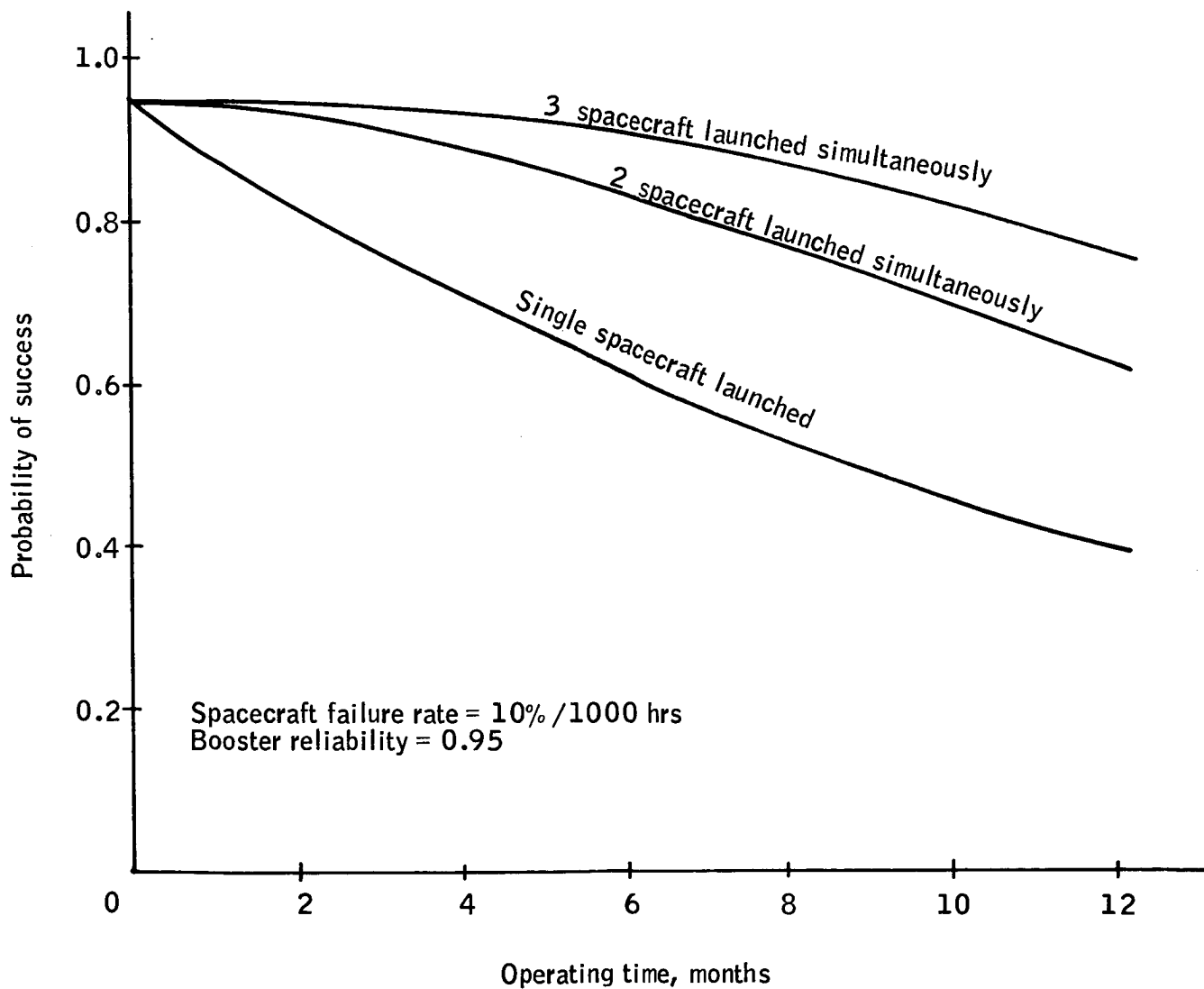


Figure 12. Variation in Probability of Success with Operating Time

There are a number of practical disadvantages to simultaneous launches. Two of these disadvantages are that a single booster failure causes loss of all spacecraft, and simultaneous launch commits all spacecraft at once, eliminating the possibility of changing the spacecraft configuration or orbit on later launches to take advantage of data gained during early flights.

Scheduled launches. - Another and more practical method for employing the reserve spacecraft would be to schedule launches either at specific intervals throughout the year or in event of in-flight failures. The interval could be selected to maximize the probability of mission success for a given number of spacecraft.

One operational approach is illustrated in Figure 13 wherein a second spacecraft is launched at the end of two months and a third at the end of eight months. The probability of at least one spacecraft surviving throughout the year is greater than 80 percent.

Assuming that random failures dominate the spacecraft probability of success, it may be argued that if a spacecraft is operating at the end of a given number of months the probability of its surviving one more month is the same as its probability of surviving the first month after launch. Using this assumption, a second, and probably the most economical, operational approach would be to launch the first spacecraft with subsequent launchings of reserve units dependent upon failure of the currently operational spacecraft. A disadvantage to this approach, however, results from the minimum launch readiness time of the launch vehicle. As mentioned previously, this period is nominally 30 days for the Delta vehicle. Therefore, under this operational concept, a failure in the orbiting spacecraft would cause a minimum 30-day interruption in data flow before a reserve unit could be launched and become operational.

Further consideration of the launch operations problem suggests still another approach which could combine the scheduled launch and failure-induced launch approaches. Examination of Figure 13 shows that if a second spacecraft is launched at the end of the second month, the probability that at least one is operational does not fall below 80 percent until the beginning of the ninth month. At this time, it might be decided in the interest of economy to withhold launching of the third spacecraft until one of the two orbiting units has failed, as the probability of one of the two continuing to function is not less than 70 percent. In the event of failure, the third spacecraft could be launched within 30 days with the probability of data interruption only 12 percent since the surviving spacecraft has an 88 percent probability of functioning for an additional month. In the event of no failures in the first two spacecraft, the third unit could be launched to extend the data gathering period to include yearly variations.

Another alternative, which could apply to all of the three discussed approaches, would be to launch the sequential spacecraft into orbit(s) other than the preferred 3:00 p. m./3:00 a. m. orbit to obtain additional data points for extracting diurnal information.

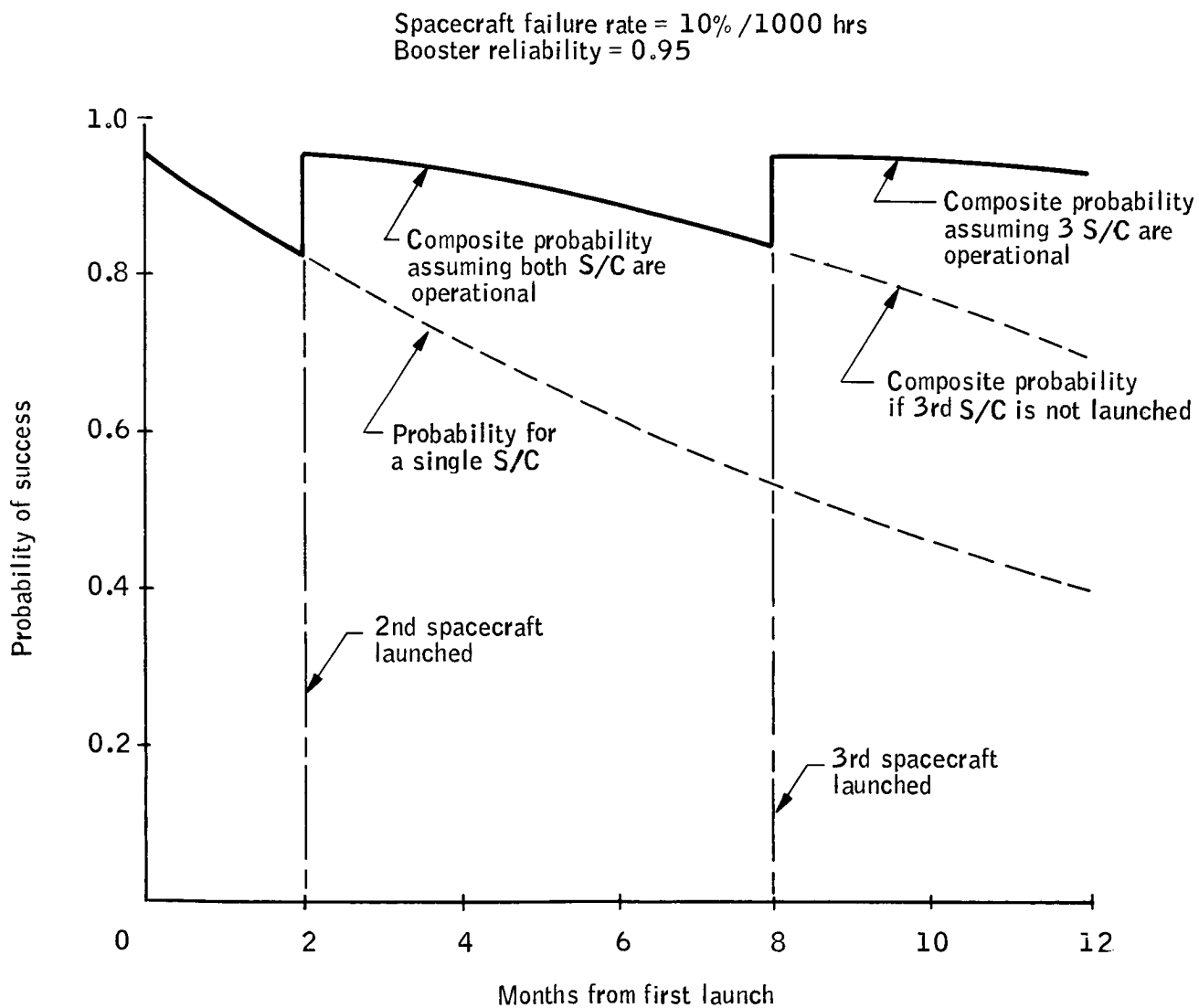


Figure 13. Effect of Scheduled Launches on Probability of Success

Recommended operational concept. - Based upon the assumption that the achievable HDS spacecraft failure rate will be approximately 10 percent/1000 hours as per the preceding examples, it is recommended that a second spacecraft be launched in a sun-synchronous orbit one to two hours later than the initial spacecraft within three months after the successful orbiting of the first spacecraft. If a failure of one of the first two spacecraft occurs in the first four or five months, a third spacecraft should be launched into the same orbit as the failed spacecraft. This approach maximizes the probability of obtaining a continuous one-year set of data while providing a 4-point diurnal data fit and also provides a reasonable probability of collecting a significant amount of data in the second year.

If the two initial spacecraft both survive the first four or five months and no anomalies appear in preliminary analysis of the data, it is recommended that the third spacecraft be launched into a still later (e.g., 6:00 to 7:00 o'clock) sun-synchronous orbit. In this way data for a six-point diurnal fit can be obtained for at least a short time, and the probability of continuous data for one year and beyond is enhanced.

Scout Compatibility

Previous cost comparisons have dealt only with the Delta vehicle. However, in view of the possible importance to the HDS mission of minimizing the interruption in data flow due to an orbiting spacecraft failure, an operational concept involving the Scout vehicle because of its 7 to 10 day readiness time, has been considered even though the present Scout vehicle is not compatible with the HDS spacecraft concept. A cursory study of an Improved Scout launched spacecraft was performed to define the basic requirements for a Scout booster improvement and to evaluate the basic mission and program tradeoffs between Delta and Scout.

The results are summarized in Figure 14, and the significant parameters are discussed below.

Configuration. - The basic Scout Launch Vehicle is shown in Figure 15 as Configuration A. An improved version, Configuration B, is shown in which the first stage thrust-to-weight is increased to provide a payload of 395 pounds into a 270 nautical mile, circular, polar orbit. If new guidance is required for improved accuracy, the payload weight capability would be reduced to approximately 340 pounds. This payload capability must further be reduced 10 to 15 percent to account for inert mechanizations to couple the HDM spacecraft to the Scout fourth stage. If additional spacecraft or payload volume is required, then the performance capabilities are given in Configuration C.

HDS booster requirements. - For the mission requirements to be met with a reduced lifetime spacecraft, the basic Scout vehicle must have an improved first stage. Guidance and control on the fourth stage may be required if the design operational spacecraft lifetime exceeds 150 days.

Parameter	Thor/Delta	Improved Scout
HDS operational lifetime	One year	Six months
HDS failure rate basis, %/1000 hr	10	15
HDS weight, lbs	600	<400
Orbit		Twilight, near circular
Diurnal coverage	Yes	No
Number of boosters	4	7
Booster reliability	0.90	0.95
Effective launch reaction time, days	30	7
Probability of achievement	90%	90%
Vehicle availability	Now under development, 1968 operational	Study only/review funding
Injection accuracy	Good	Marginal to satisfactory
Volume constant	Existing 57" heat shield	Existing 34" diam heat shield
Growth capabilities	Excellent	Poor
Data confidence	Good	Better
Flexibility & operational control	Poor	Good
HDS subsystem		
Experiment package	Redundant	Nonredundant
Radiometer	Cooled detector	Cooled detector
Cooler	Solid cryogen	Reduced size solid cryogen
Structure	Hexagonal cylinder	Hexagonal cylinder

Figure 14. HDS Program Alternatives with Delta and Scout

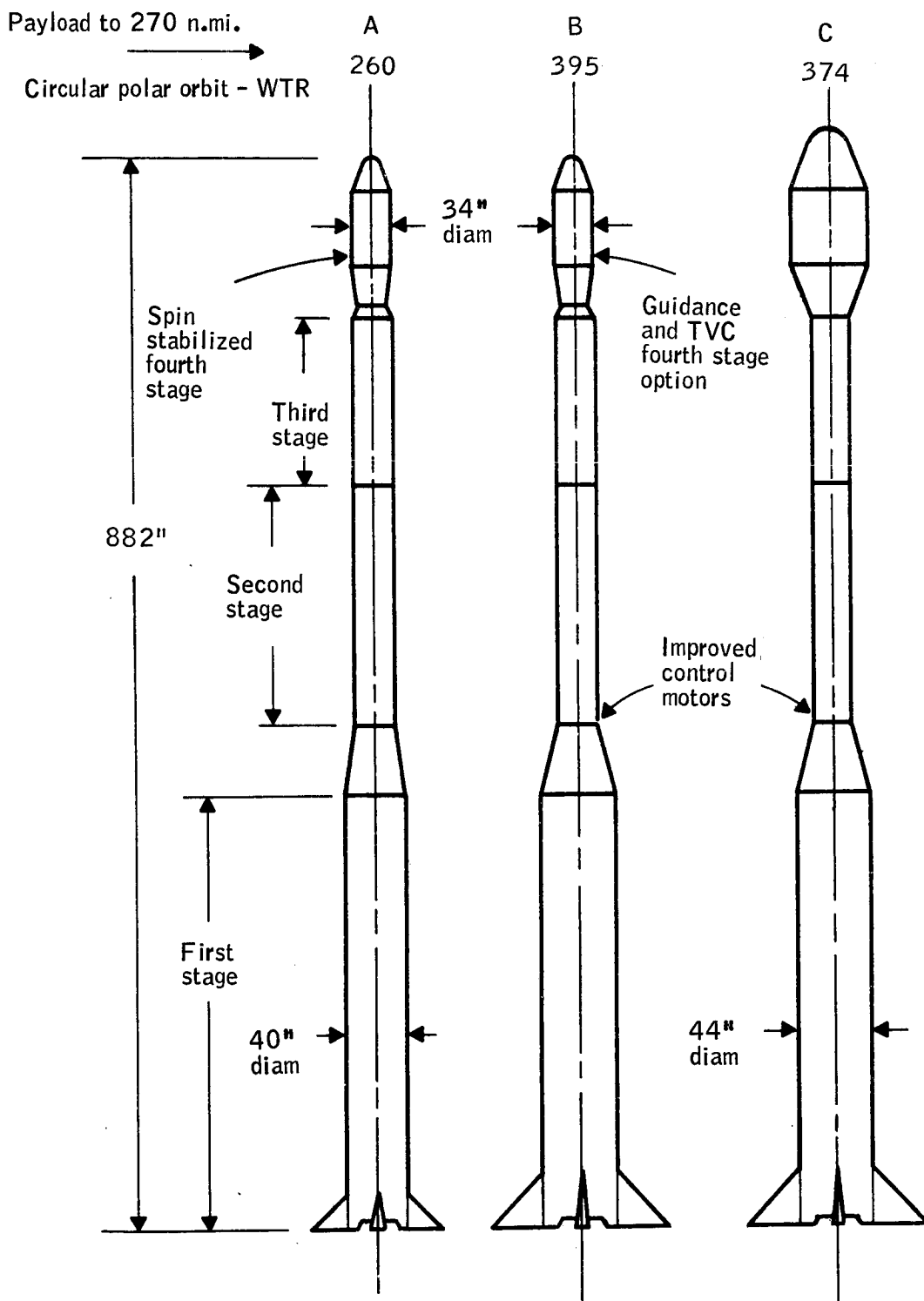


Figure 15. Scout Growth

Orbital injection accuracy. - The estimated inclination and altitude errors are listed in Table 2. These errors result in orbital precession rate variations that constrain the solar cell power system design as shown in Figure 16. Here the sun-line to orbit-normal angle is plotted against days from launch. The existing or spin stabilized fourth stage Scout shows excessive excursions about the nominal beyond 150 days from launch, while the two-stage Delta is satisfactory throughout the year. Thus, the Scout vehicle guidance system must be improved to meet the mission orbit injection accuracy requirements.

TABLE 2. - NOMINAL INJECTION ACCURACIES,
3-SIGMA VALUES - 270 NAUTICAL
MILE CIRCULAR ORBIT

Error	Delta			Scout		
	2 Stage	3 Stage		66	Study	
		Spin	Guided		Spin	Guided
Inclination, degrees	0.18	0.6	0.18	0.64	0.64	0.18
Apogee to perigee, nautical miles	30	45	45	73	73	73

Spacecraft heat shield. - The existing 34-inch diameter heat shield is shown in Figure 17. The heat shield is of fiberglass honeycomb sandwich construction with a stainless steel nose cap. It is designed to carry fourth stage vehicle loads and to keep the payload compartment temperatures within specified limits. A bumper between the heat shield and motor keeps the spacecraft from hitting the heat shield under high load conditions and transfers lateral inertial loads from the motor case to the heat shield.

Increasing the heat shield diameter above 34 inches would require a detailed structural,dynamic analysis. However, the 48-inch diameter bulbous Delta fairing may possibly be made compatible with the Improved Scout. Any Scout-launched HDS concept considered in this study would utilize the existing 34-inch diameter heat shield.

Reduced lifetime HDS Scout launch vehicle. - An analysis was made of the comparative cost effectiveness of an HDS spacecraft compatible with the Scout launch vehicles to provide a one-year mission data profile. The parameters considered were:

- 1, 1/2, 1/3, and 1/4 - year spacecraft operational lifetime.

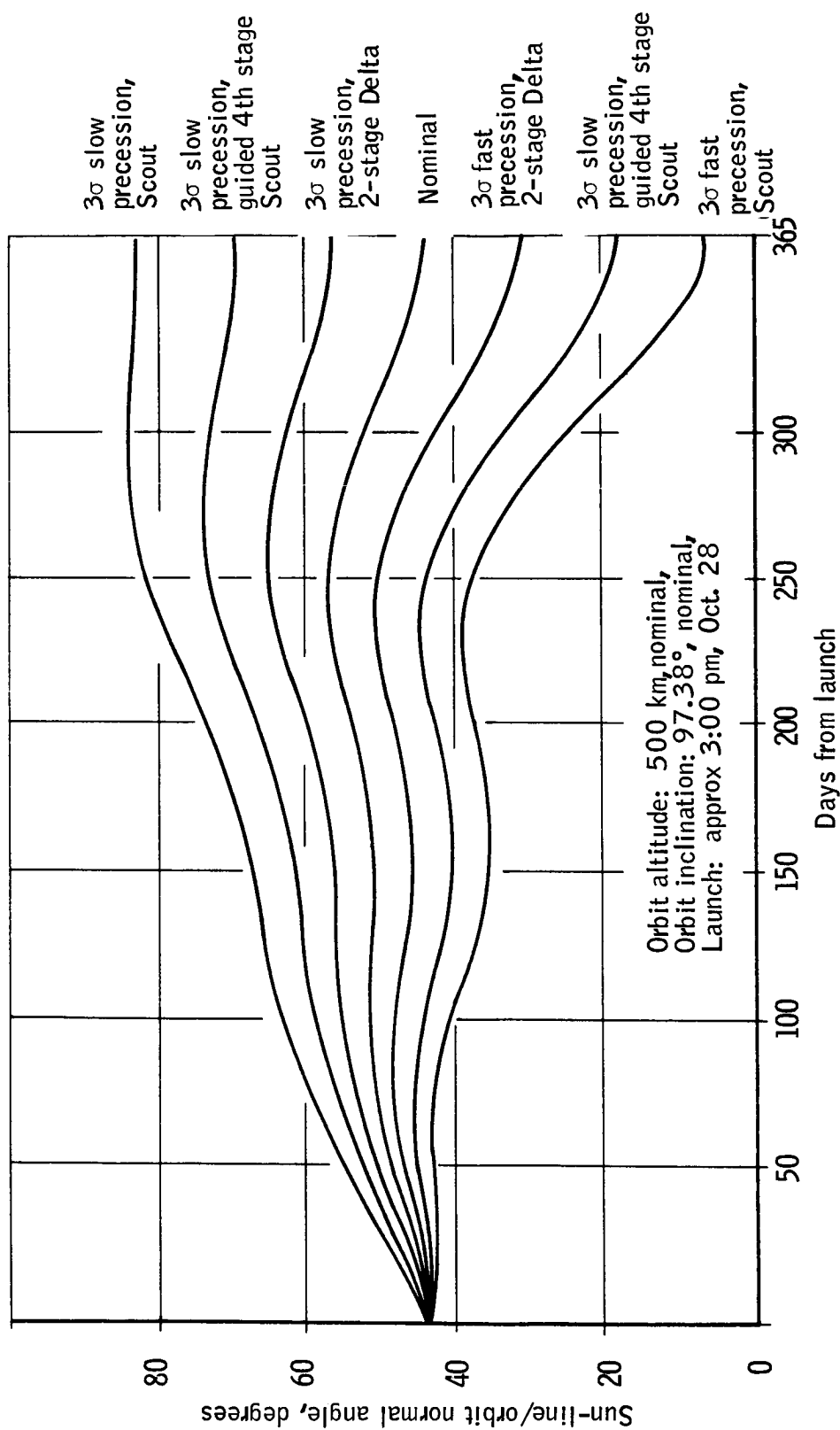


Figure 16. Sun-Line/Orbit-Normal Angle versus Days,
3 O'Clock, Sun-Synchronous Orbit

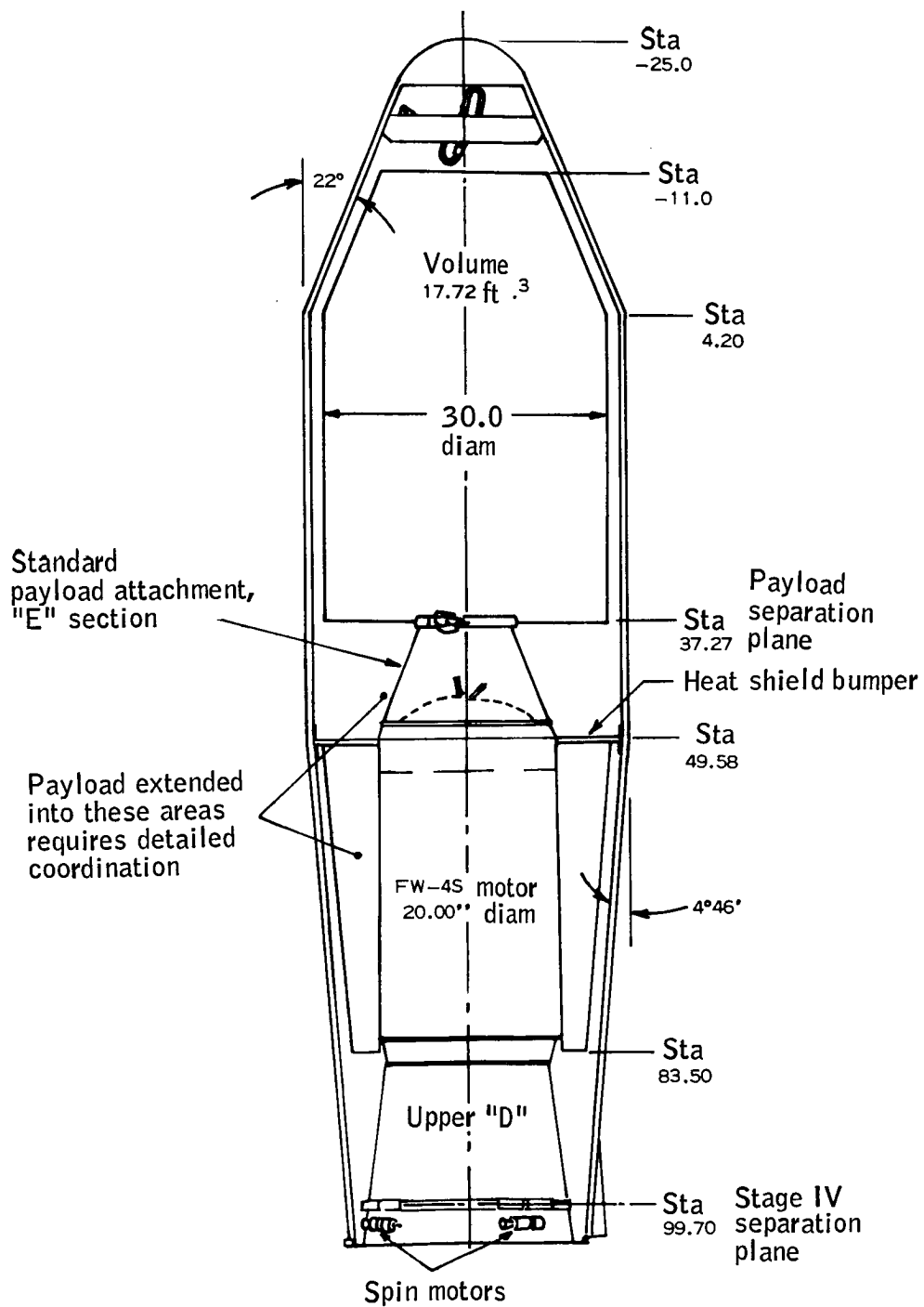


Figure 17. Scout Payload Envelope

- Spacecraft failure rates from 10 to 30 percent per 1000 hours.
- Booster reliabilities $0.85 < R_B < 0.95$
- Nominal injection accuracies 2-stage Delta and 4-stage (spin stabilized fourth stage) Scout
- Booster launch readiness - 5 to 90 days.

The significant results are shown in Figure 18. This figure indicates that lifetimes less than six months with Scout are not comparable in cost effectiveness with one-year lifetimes on Delta. This is due to the initial inventory and standby costs for hardware. This comparison does not include development costs which should be approximately equal for either alternative.

The comparative weight summaries given in Table 3 represent the current system as defined for the Delta vehicle and a very basic subsystem weight allocation based on the Scout launch vehicle capabilities. The Scout spacecraft weight reflects no redundancy in the experiment package, nominal spacecraft operational life of six months with a corresponding reduction in cryogenic cooler weight, and a sun-synchronous twilight orbit for electrical power system efficiency. The experiment package concept, as defined in this study and including the requirement for shielding the radiometer view port, is not compatible with either the Scout vehicle orbital weight restrictions or the current Scout payload fairing envelope. An HDS system compressed to fit the Scout launch vehicle would require such drastic compromises of the basic measurement accuracies as to render such a system extremely uneconomical in terms of extending current knowledge of the earth's horizon.

TABLE 3. -ESTIMATED WEIGHT SUMMARY

Subsystem	Scout	Delta
Experiment package	160	324
Power supply	40	121
Attitude control	29	48
Communications	25	26
Data handling	45	59
Structures	70	145
Total weight	369	723
Lifetime	6 months	12 months

LAUNCH VEHICLE CONFIGURATION AND CONSTRAINTS

The following paragraphs describe the Delta launch vehicle and constraints on spacecraft design imposed by the launch vehicle characteristics. The basic

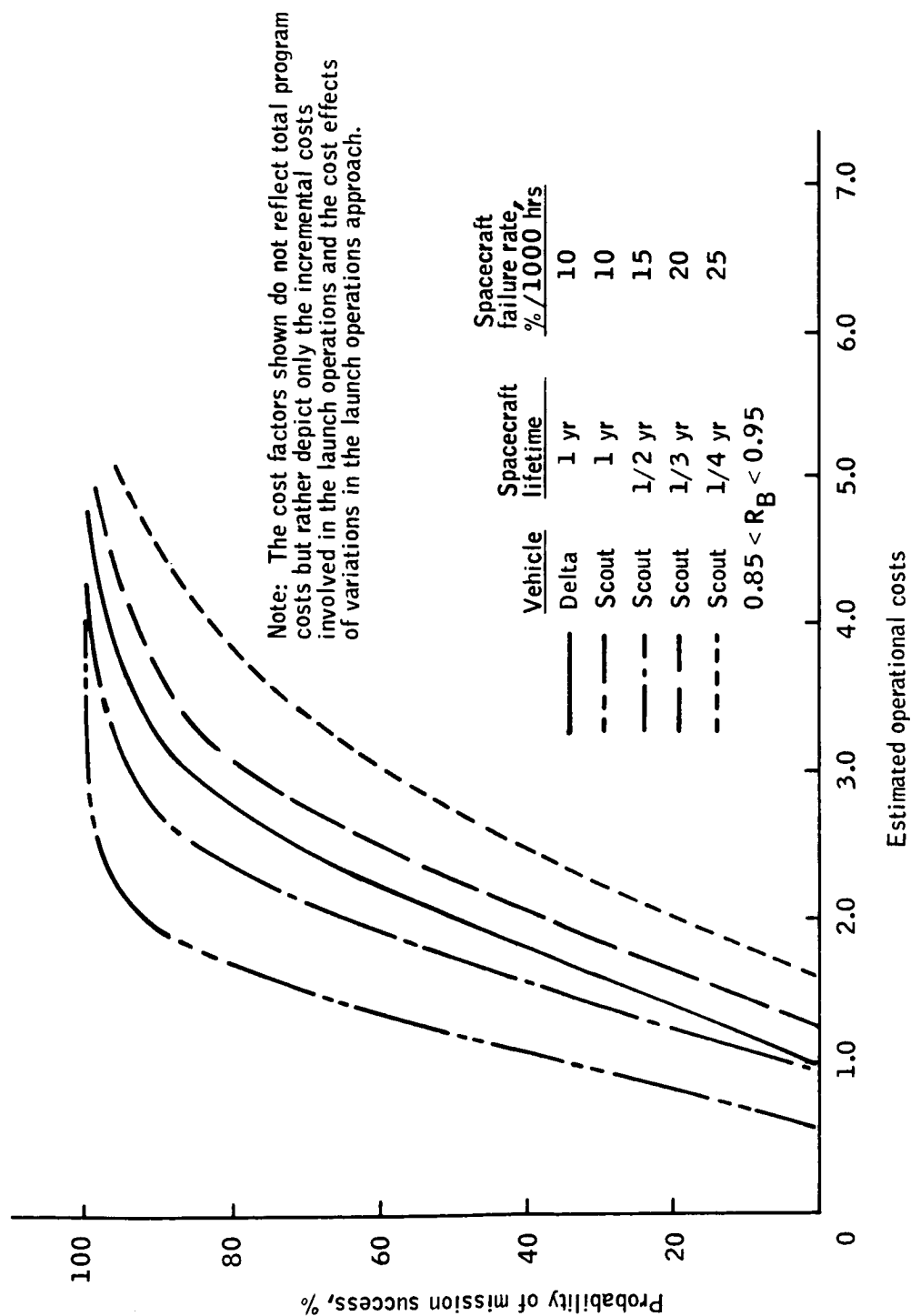


Figure 18. Comparative Launch Costs and Performance Envelope Delta and Scout Vehicles.

information was extracted from the Improved Delta Spacecraft Design Restraint Document, revision of January 1966, prepared by Douglas Aircraft Company. The description therein was updated to include modifications and revisions which are now being developed and will be incorporated in the launch vehicle during the operational time period of the HDS mission.

Description

The launch vehicle is shown in plan view in Figure 19. The first stage is a modified, thrust-augmented Delta (TAD) booster.

A cylindrical first-to-second stage adapter section is attached to the forward end of the first-stage transition section. First-to-second stage separation occurs at the forward face of the adapter section. The second-stage vehicle consists of two major components - a liquid propellant propulsion system and guidance and equipment compartment structure - and protects the spacecraft from aerodynamic pressures and heating during launch.

The first-stage liquid-propellant booster is powered by a gimballed main engine and initially augmented by three externally mounted solid-propellant motors equally spaced around the external periphery. The three solid-propellant motors are jettisoned simultaneously after solid-propellant burn-out. The second stage is a pressure-fed propulsion system. The thrust chamber assembly is mounted on a gimbal system for attitude control (pitch and yaw) during powered flight. Roll control during powered and coast flight and pitch and yaw control during coast flight is achieved by the second-stage cold gas system. This system provides the capability for payload orientation following orbital injection.

The second-stage guidance compartment structure houses the flight control, radio guidance, velocity cutoff, range safety, tracking, power systems, and diagnostic instrumentation. A spin table and spacecraft attach fitting are mounted on the forward end of the guidance compartment structure. The spacecraft attach fitting includes provisions for separating the spacecraft from the second stage.

Spacecraft Design Restraints

Vibration. - The vibration restraints for spacecraft to be launched on an Improved Delta vehicle are based on NASA GSFC vibration qualification tests to be performed on prototype spacecraft and acceptance vibration tests on flight hardware. The acceptance test criteria represents flight equivalent vibration levels combined with a test factor.

In flight, the spacecraft will be subjected to both sinusoidal and random vibrations. During first- and second-stage operation, maximum random vibrations occur during liftoff and transonic flight. In addition, low frequency sinusoids will be superimposed on the random vibration at liftoff. During the last 25 seconds of first-stage operations, a sinusoidal oscillation varying between 17 and 25 cps is experienced throughout the vehicle.

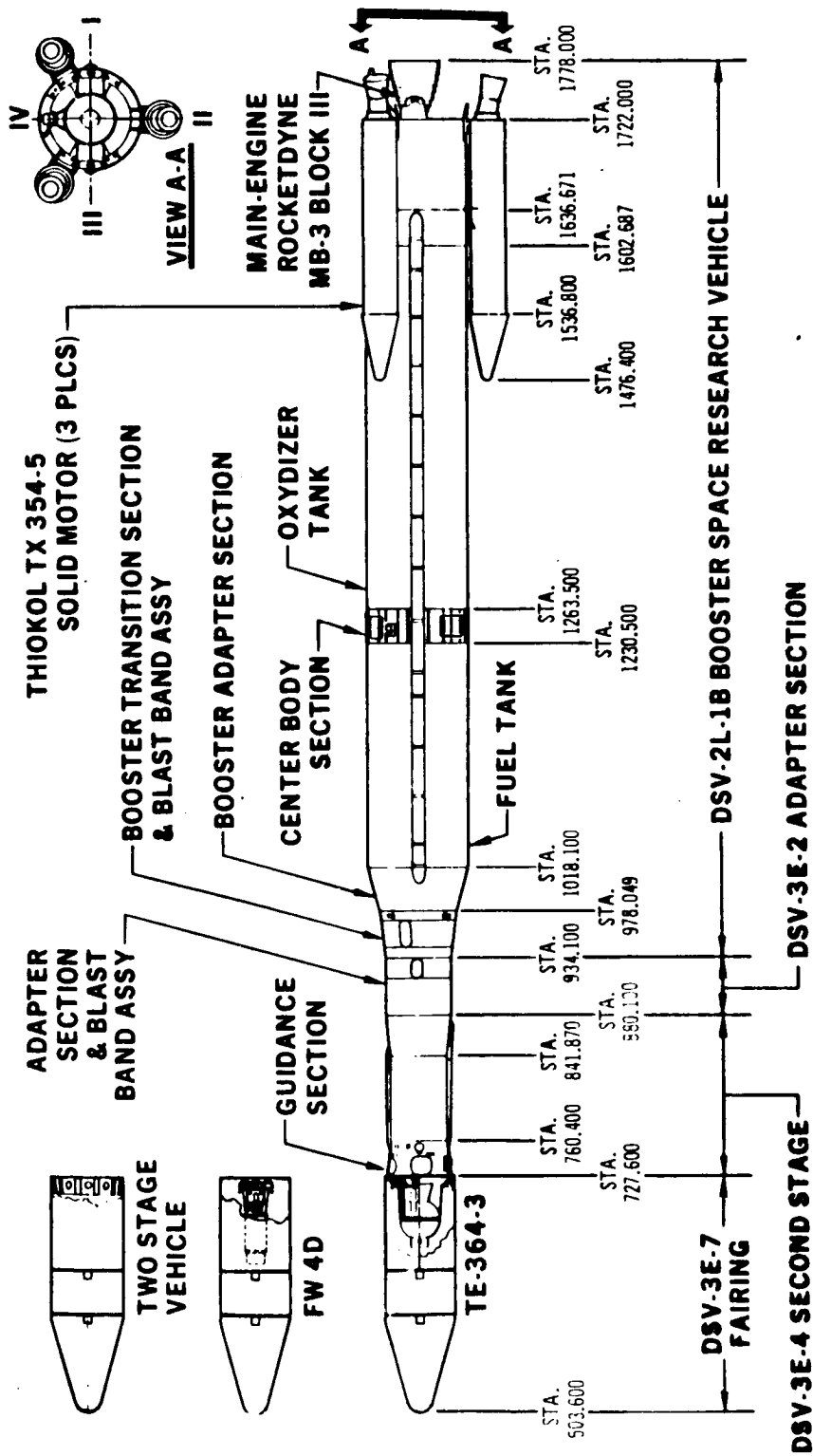


Figure 19. Delta Launch Vehicle

Table 4 shows sinusoidal and random vibration restraints for the DSV-3N vehicle. The data was taken from an unpublished revision to the NASA GSFC "General Environmental Test Specification for Spacecraft and Components".

Acceleration. - Estimated flight accelerations for the DSV-3N vehicle are as follows:

Condition I (liftoff)	3 g forward combined with 2 g in any lateral direction
Condition II (1st stage 95% burn)	8 g forward combined with 2 g in any lateral direction

TABLE 4.- DSV-3N SPACECRAFT VIBRATION RESTRAINTS

Flight sinusoidal vibration restraints				
Axis	Frequency, cps	Level, g, 0-peak (a)	Sweep rate (b)	
Thrust (Z-Z)	10 - 16	2.0	2 octaves/minute	
	16 - 26	2.5		
	26 - 250	2.0		
	250 - 400	5.0		
	400 - 2000	10.0		
Lateral (X-X) and (Y-Y)	5 - 250	1.5	2 octaves/minute	
	250 - 400	5.0		
	400 - 2000	10.0		
^a Qualification levels = 1.5 x flight levels ^b Qualification sweep rate = 1/2 flight equivalent rates				
Flight random vibration restraints				
Axis	Frequency, cps	PSD level, g ² /cps (a)	Acceleration, g (rms)	Duration, minutes (b)
Thrust (Z-Z)	20 - 150	0.01	12.3	1 min/axis
	150 - 425	+4 dB/octave	12.3	
Lateral (X-X)	425 - 1200	0.04	12.3	
Lateral	1200 - 2000	-2 dB/octave	12.3	
^a Qualification levels = 2.25 x flight levels ^b Qualification duration = 2 x flight duration				

Thermal. - The spacecraft fairing provides protection from aerodynamic heating effects during launch through the atmosphere. As an example of the temperatures to which the spacecraft might be subjected, Figure 20 presents a time history of the temperature at various locations on the fairing during a maximum heating trajectory launch.

The fairing is of fiberglass construction, and, in order to prevent spacecraft contamination resulting from outgassing of the phenolic resin in the fiberglass, the fairing interior surface must be held below approximately 450°F ($\approx 500^\circ\text{K}$). This can be accomplished by the application of external insulation. Additional protection for sensitive areas of the spacecraft can be provided by internal insulation.

Spacecraft orientation separation errors. - The HDS spacecraft must be oriented with its spin axis normal to the orbit plane following orbital injection. The Delta DSV-3N vehicle is planned to provide this capability through the second-stage attitude control system. The accuracy with which the orientation maneuver can be accomplished imposes a design restraint upon the spacecraft attitude control system. In the absence of quantitative supporting data, the NASA GSFC Delta Project Office estimates that the desired orientation can be accomplished to an accuracy of ± 1 degree.

An additional spacecraft control system design restraint follows from the possible errors in attitude resulting from the separation maneuver. Separation from the second stage is afforded through a system of matched springs. Practical limitations in matching the spring characteristics allow a possible tip-off error of 3 degrees/second.

Spacecraft/launch vehicle interface. -

Fairing: The spacecraft is covered by a transparent fairing to protect it from aerodynamic and thermodynamic effects while in the atmosphere. The fairing is jettisoned during second stage powered flight at a minimum altitude of 60 n. mi.

The standard fairing is 224 inches long and 65 inches in diameter. Without insulation it weighs approximately 535 pounds. Figure 21 shows the fairing structural envelope within which the spacecraft dimensions must be fitted.

The fairing is constructed in two separate half-shell sections which surround the spacecraft and are clamped together with two strap assemblies at the forward end and a frame at the aft end. Tension in both straps and in the frame is affected by fittings held together with explosive bolts. In addition, two spring-loaded nose latches inside the fairing hold the fairing halves together.

At fairing ejection, all explosive bolts are broken simultaneously, the latches open, and a pair of preloaded spring cartridges, mounted forward of the cylindrical section, act to thrust the half-shells away from the parting plane. The half shells then rotate about a pair of pivot fittings on the aft frame. A pair of separation springs at the parting plane effect an even rotation about the pivot fittings. A typical fairing trajectory is illustrated in Figure 22.

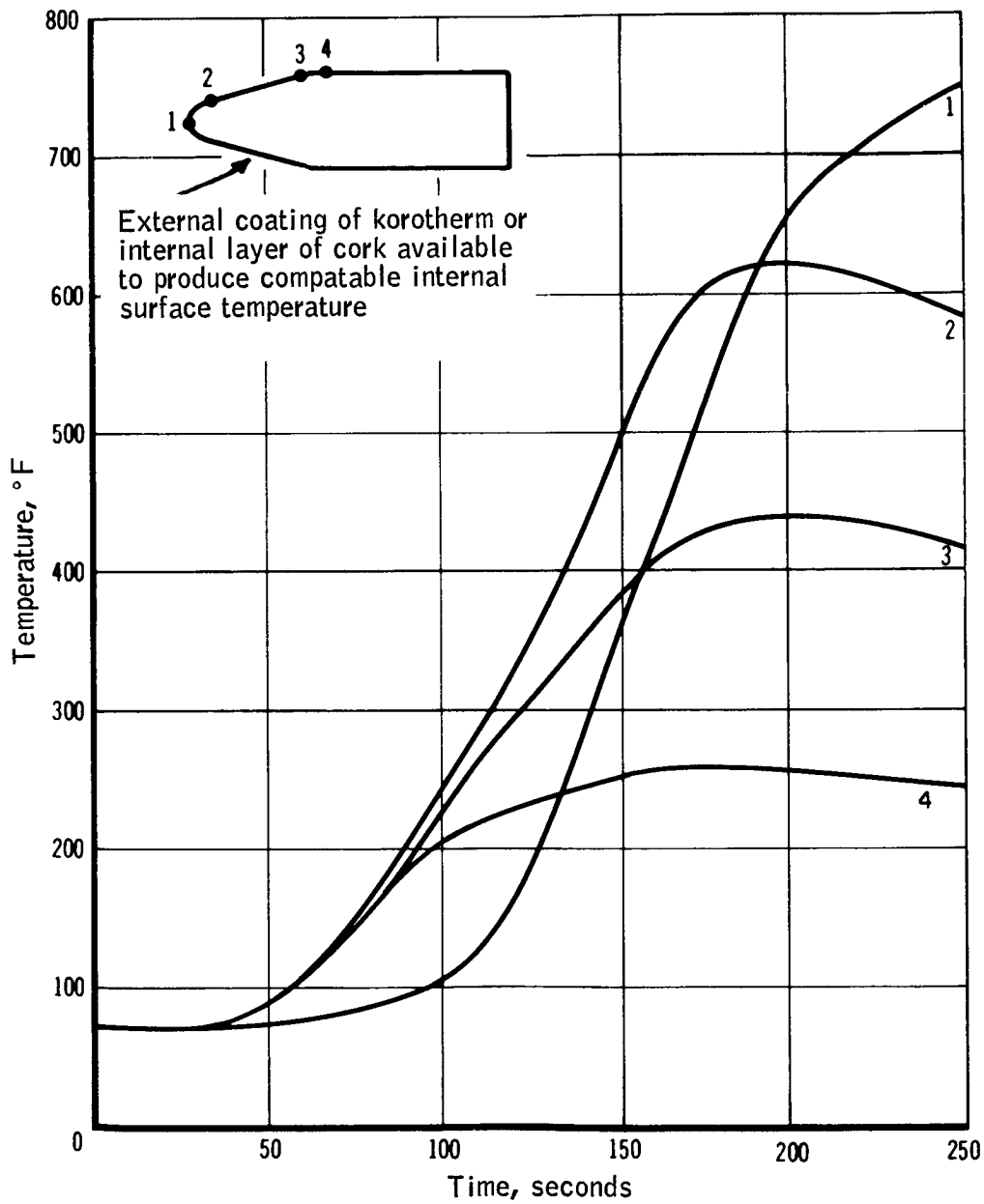


Figure 20. Improved Delta Fairing Internal Surface Temperature History Maximum Heating Trajectory

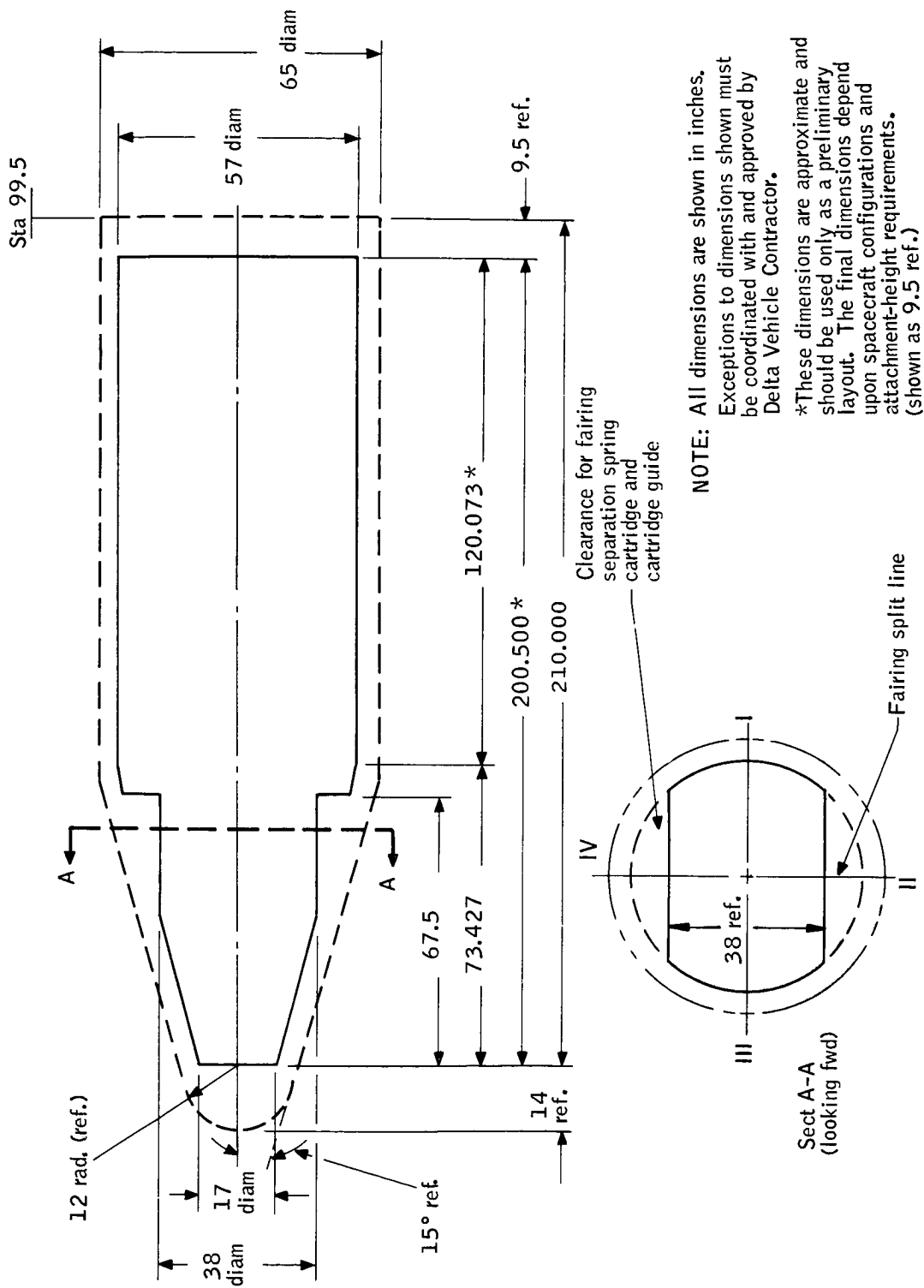


Figure 21. Improved Delta Fairing/Spacecraft Envelope, Direct Mounting on Second Stage

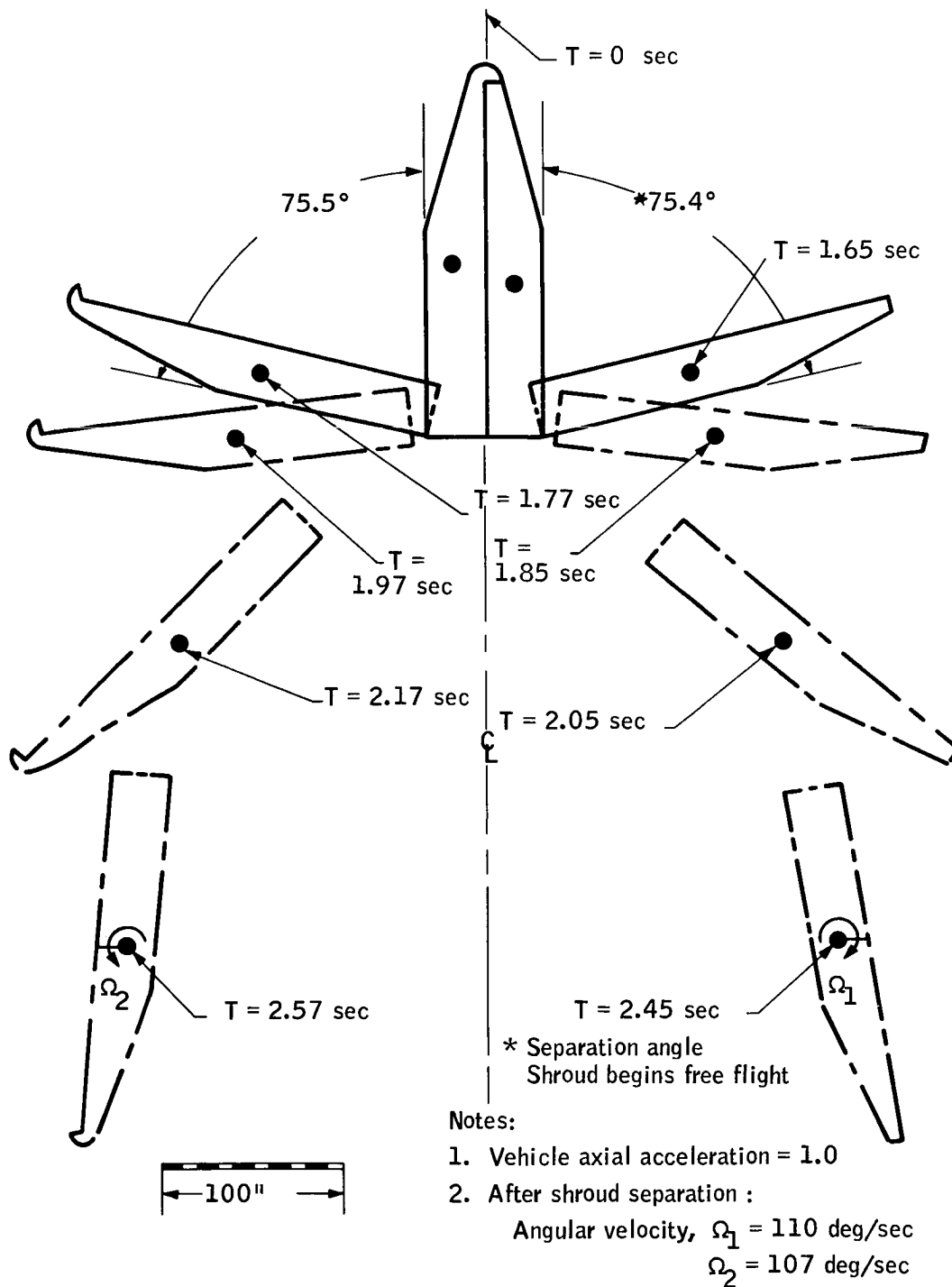


Figure 22. Typical Improved Delta Fairing Trajectory

Attach fittings: The spacecraft attach fitting is the structure which supports the spacecraft on the second-stage spin table. The spacecraft is attached to the fitting by a circular V-type clamp assembly which releases in flight by firing two explosive bolt cutters. Separation is then effected by the separation springs.

Attach fittings are provided by the launch vehicle manufacturer with generally standard diameters at 9, 18, 20, 24, and 37 inches. If possible, the spacecraft attach scheme should be planned to accommodate a standard fitting size.

Spacecraft contamination. -- All pyrotechnic devices used in the fairing and the spacecraft separation are self-contained and will not contaminate the experiment package optical systems.

FLIGHT OPERATIONS PLAN

The HDS mission operational requirements must be integrated into an overall operational plan compatible with the launch vehicle requirements and capabilities. Prelaunch operations of the launch vehicle are an established sequence of events based upon a history of successful Delta launches. Into this sequence the necessary spacecraft preparations must be fitted. Figure 23 shows the major events that occur during the operation. The three significant operational phases of prelaunch, launch, and standby are discussed below.

Prelaunch Phase

Spacecraft development and qualification proceeds in parallel with launch vehicle production. The spacecraft is delivered to the launch site in a flight-qualified status approximately 90 days prior to launch. All spacecraft systems are thereafter checked and functionally verified. The launch vehicle is delivered to the launch site approximately 30 days prior to launch. The first- and second-stage boosters are checked out individually, and the first stage is erected on the launch pad. The second stage is then mated to the first stage. Upon completion of spacecraft checkout and approximately 4 to 7 days prior to launch, the flight spacecraft is mated to the launch vehicle, and rf system tests are conducted. Spacecraft/vehicle compatibility and spacecraft systems checks are conducted from this time until approximately two days prior to launch. The tasks from this time until launch are accomplished in a pre-established sequence documented in a mission countdown manual. The significant events from the spacecraft standpoint include:

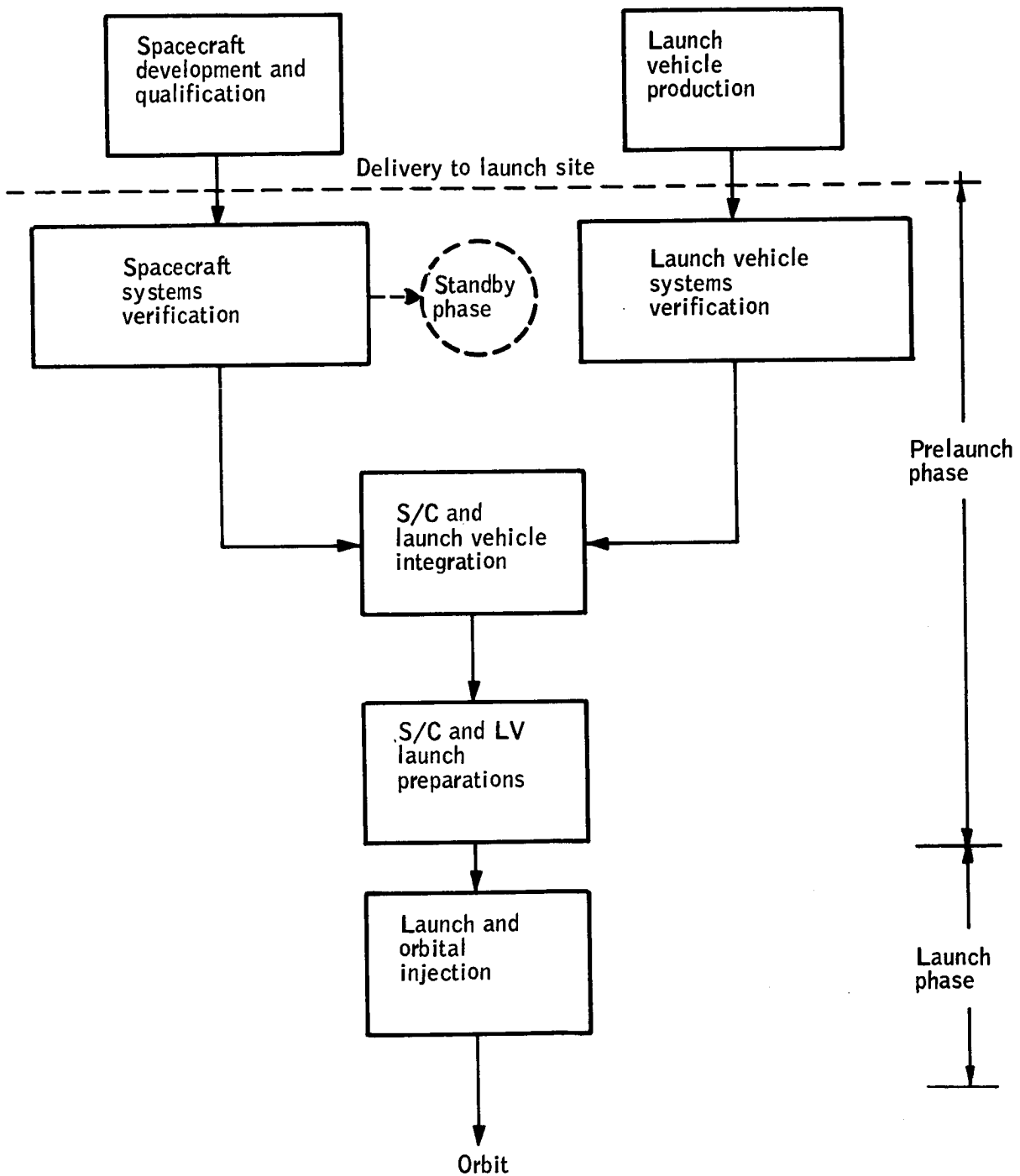


Figure 23. Spacecraft/Launch Vehicle Operational Flow Diagram

<u>Time before launch</u>	<u>Event</u>
2 days	Begin countdown tasks
1 day	Install payload fairing
3 hours	Remove gantry
1-1/2 hours	Spacecraft final checks
20 minutes	Final countdown

Launch Phase

HDS mission requirements specify that the spacecraft be launched into a near polar, circular orbit at an altitude of approximately 270 n. mi. and an inclination of 97.38 degrees. The orbit characteristics were chosen to produce an essentially sun-synchronous orbit and a minimum spacecraft orbit lifetime of one year.

The spacecraft is a spin-stabilized vehicle with its spin axis normal to the orbit plane. In this attitude the spacecraft appears to roll like a wheel while in orbit.

During launch the spacecraft is mounted on the second-stage vehicle with its spin axis parallel to the thrust axis. At the orbit injection point, the spin axis is therefore parallel to the orbit plane, and a 90-degree yaw maneuver of the launch vehicle is required for proper orientation.

Following the orientation maneuver, the spacecraft solar arrays must be deployed and the spacecraft operational spin rate (3 rpm) established prior to separation. The sequence of events is illustrated in Figure 24.

The planned Delta launch vehicle will provide the orientation maneuver capability through a preprogrammed sequencing of its cold gas attitude control system. The presently planned launch vehicle second-stage spin table must be modified to accommodate the low spin rate necessary for the spacecraft. An electric or pneumatic motor could provide this capability.

With a tip-off rate of three degrees per second and a spin rate of three rpm (as determined in the attitude control system analysis), the spacecraft will be precessing with a half-cone angle of eight degrees after separation from the booster. One full orbit has been allowed for the damping of this coning motion, and the proposed damping system can adequately correct the attitude to the desired 0.5 degree half-cone angle in this period. The tip-off rate of three degrees/per second is the design constraint published by the Douglas Company for matched-spring separation systems used on the Improved Delta.

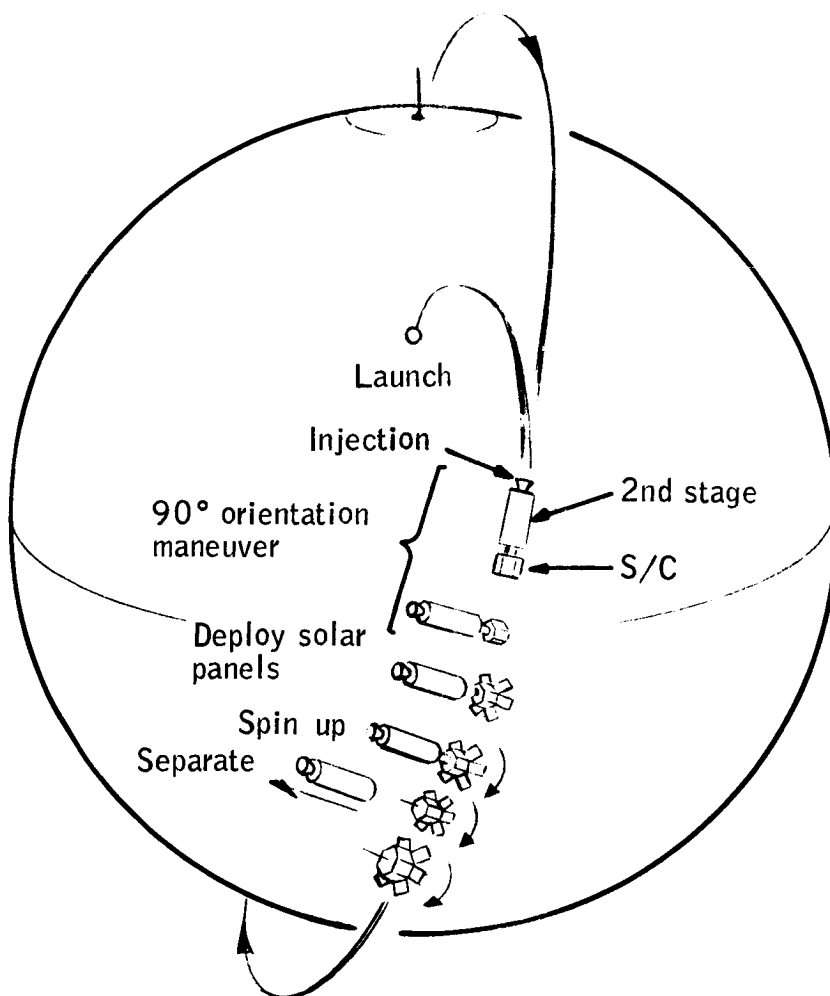


Figure 24. Launch Phase Events

A simulation of the HDS mission ascent trajectory was conducted by the Douglas Missile and Space Systems Division using predicted Delta DSV-3N vehicle inputs. From this exercise the launch profile, ground track, nominal vehicle attitude, and estimated orbit errors were determined. They are presented in Figures 25 and 26 and Table 5.

TABLE 5. - HDS MISSION ESTIMATED ORBIT ERRORS

Injection point parameters	Nominal	One-sigma deviation
Flight path elevation	-0.0023 deg	0.10
Flight path azimuth	187.74 deg	0.16
Vehicle centerline elevation	-10.06 deg	1.7
Vehicle centerline azimuth	189.61 deg	0.3
Inclination	97.42 deg	0.04
Altitude	269.9 n. mi.	2
Pertinent trajectory parameters		
Launch vehicle	DSV-3N	
Payload	670 lbs (incl 70 lbs for attach fitting)	
Launch site	Western Test Range	
Initial flight azimuth	191 degrees	
Ascent mode	Direct injection	

Operational Concept for Multiple Flights

The recommended operational concept evolved during this study is the scheduled, sequential launching of two spacecraft with the third spacecraft being held in reserve in the event of failure of either of the two orbiting units. If the first spacecraft functions properly, the second spacecraft will be launched into an identical orbit with the exception that nodal time would be varied to provide additional data which may be useful in defining diurnal effects. If either of the two orbiting spacecraft fails within the first six months, the third spacecraft will be launched into the same orbit as the failed unit. In the event of no failures in the first six months, the option may be taken to launch the third spacecraft into an orbit differing in nodal time from either of the others and thus gaining additional diurnal data effects.

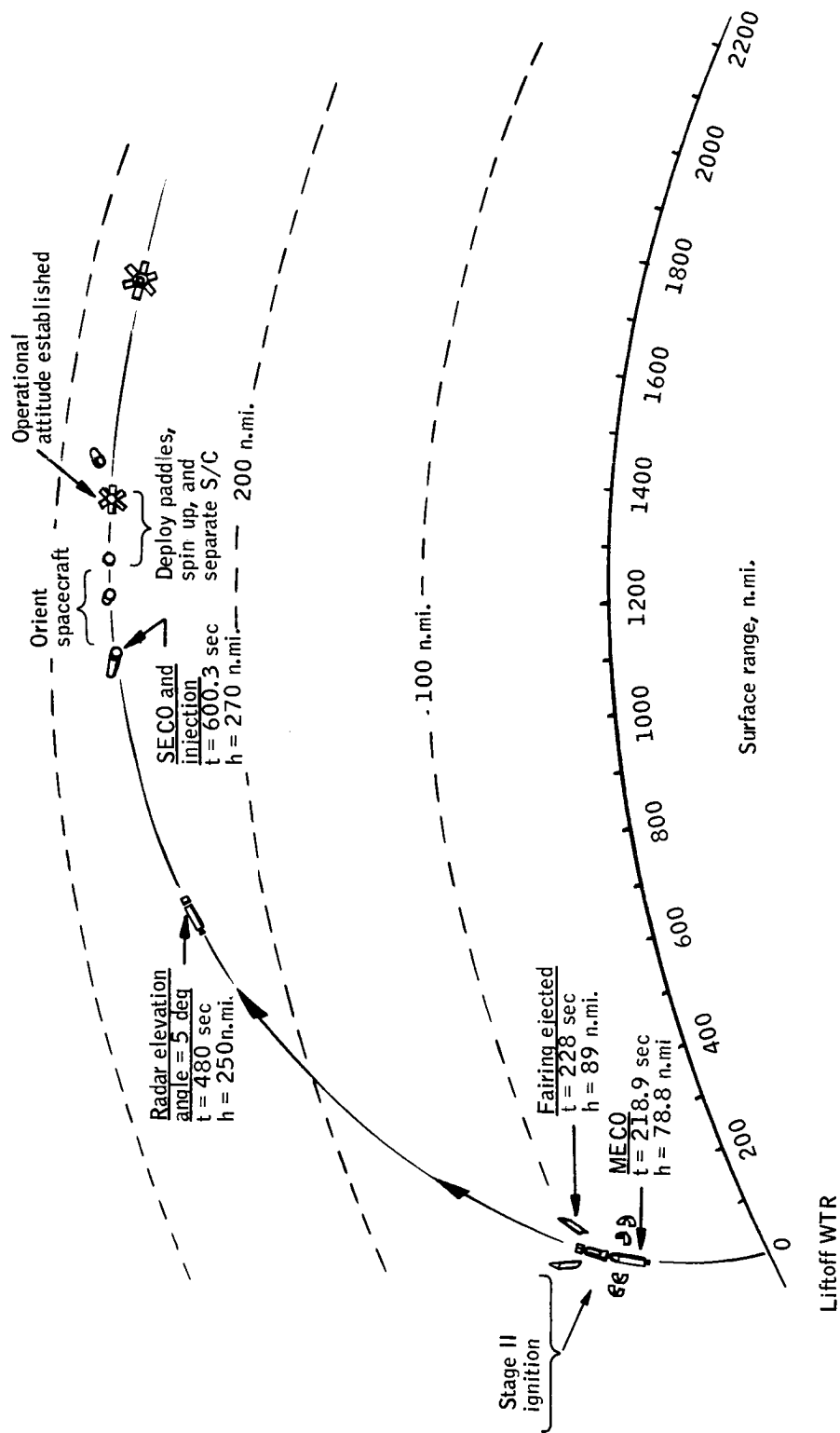


Figure 25. HDS Mission Launch Profile

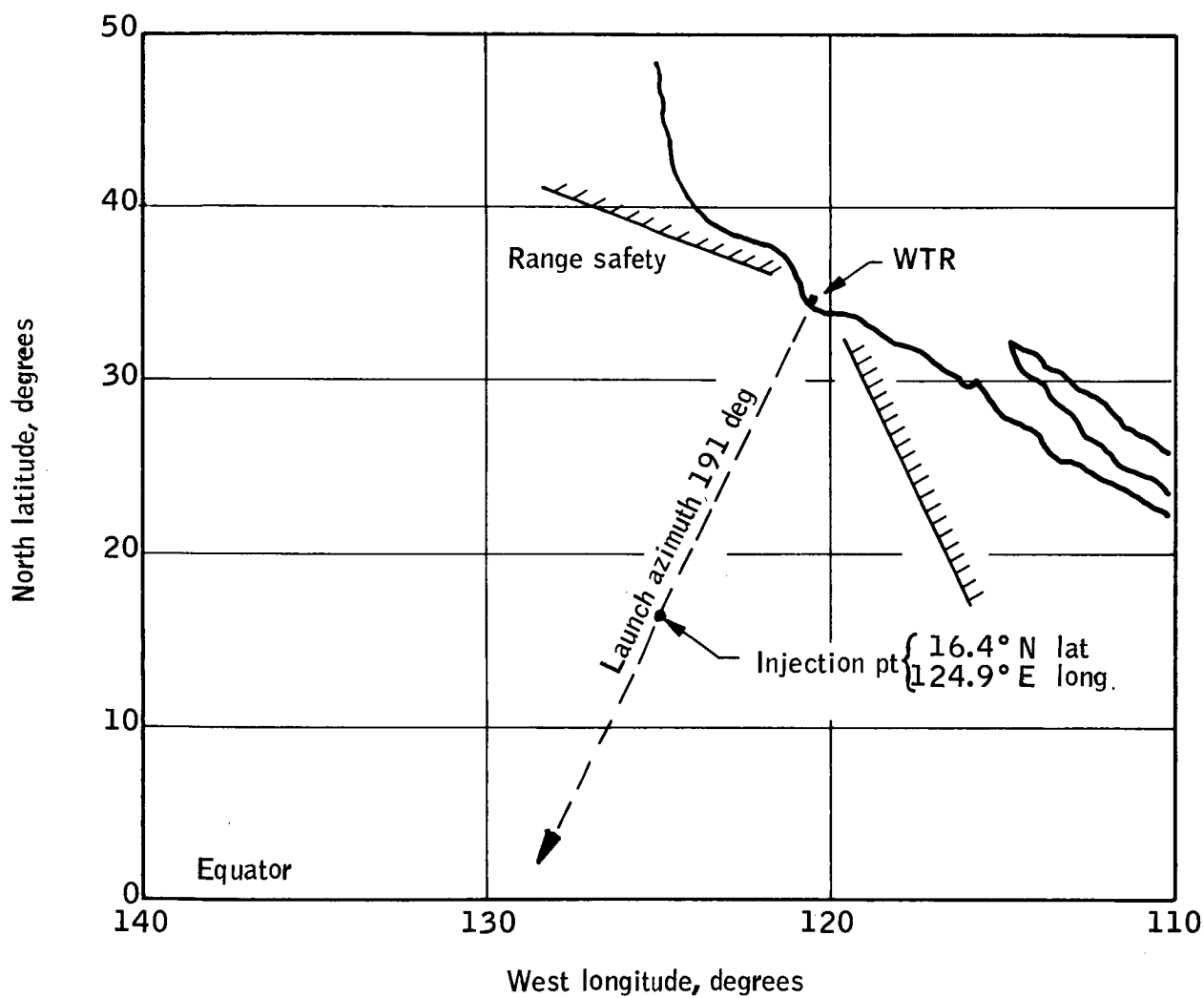


Figure 26. HDS Mission Launch Phase Ground Track and Injection Point

LAUNCH SUPPORT

The study of launch support requirements was conducted to identify the constraints imposed upon the HDS operational plan by launch site or vehicle considerations and to define for future development any special equipment required to support the prelaunch and launch operations.

LAUNCH SITE CONSTRAINTS AND REQUIREMENTS

Launch Site

The HDS spacecraft will be launched from the Western Test Range. The following is a brief general description of the on-site facilities available for spacecraft launch activities.

Spacecraft are brought to the WTR and prepared for launch in the NASA-provided spacecraft laboratory building diagrammed in Figure 27. The building is composed of three main areas: high bays, low bays, and office space. Within the building are spacecraft assembly areas, spacecraft lab areas, a clean room, rf screen room, dark room, computer facility, office space, and the NASA ULO telemetry doppler station.

Once the spacecraft is checked and prepared for launch, it is transported to the launch site (SLC-1) located approximately 12 miles from the spacecraft lab building (see Figure 28). There it is mounted on the launch vehicle, and subsequent final check-out is conducted by rf link to a 450-foot antenna tower at the spacecraft lab building.

The launch site consists of a launch control blockhouse, Delta operations building, vehicle shelters, gantry, umbilical mast, rf tower, and shelters for support equipment.

The Delta vehicle launch pad is shown in Figure 29. The first-stage booster is prepared and checked horizontally in a roll-back shelter and then erected within the 151-foot gantry shown in Figure 30. The Delta second stage is mounted on the first stage, and the spacecraft mounts on the second stage. The fourth and fifth levels of the gantry are enclosed with rf transparent fiberglass panels to provide a spacecraft work area.

The umbilical mast provides wiring from the spacecraft to a console in the blockhouse where a limited number of payload functions may be checked during the terminal countdown.

Launch Vehicle

The HDS spacecraft will be launched into orbit by an Improved Delta vehicle. The launch vehicle was described in a previous section of this report, and all launch vehicle support requirements are provided by the Douglas Launch Operations Crew.

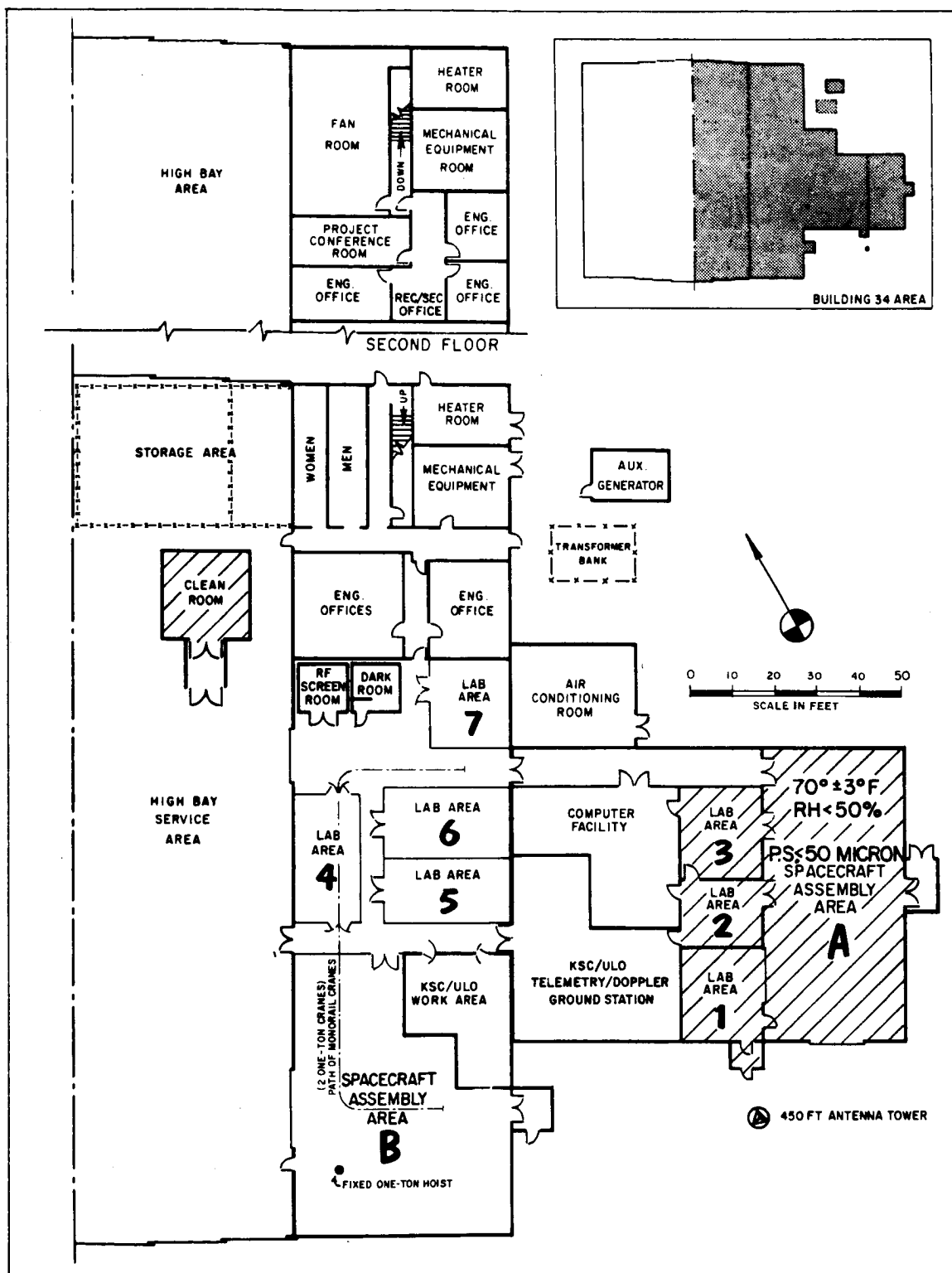


Figure 27. Spacecraft Laboratory, Building 34

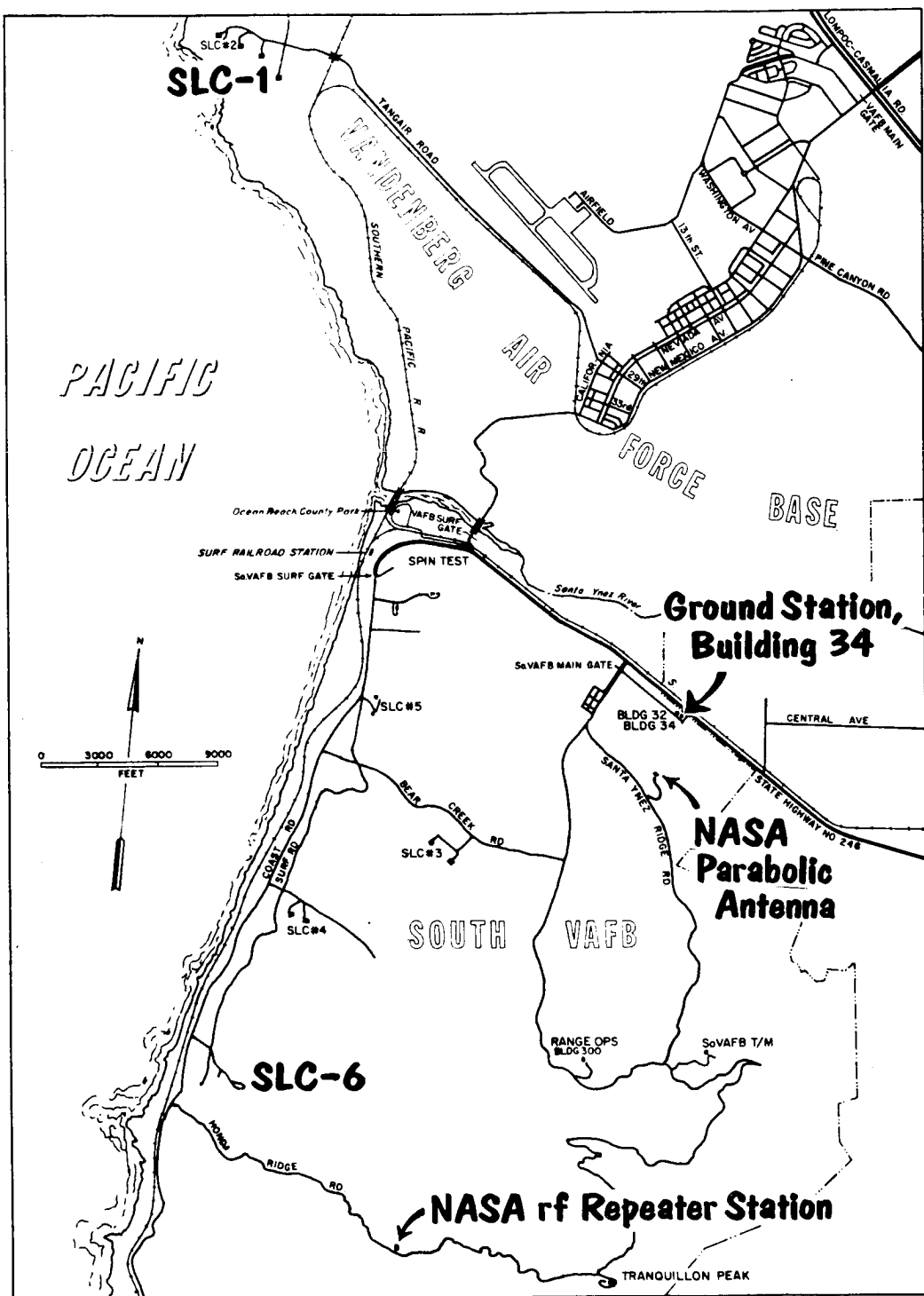


Figure 28. WTR Launch Site Location

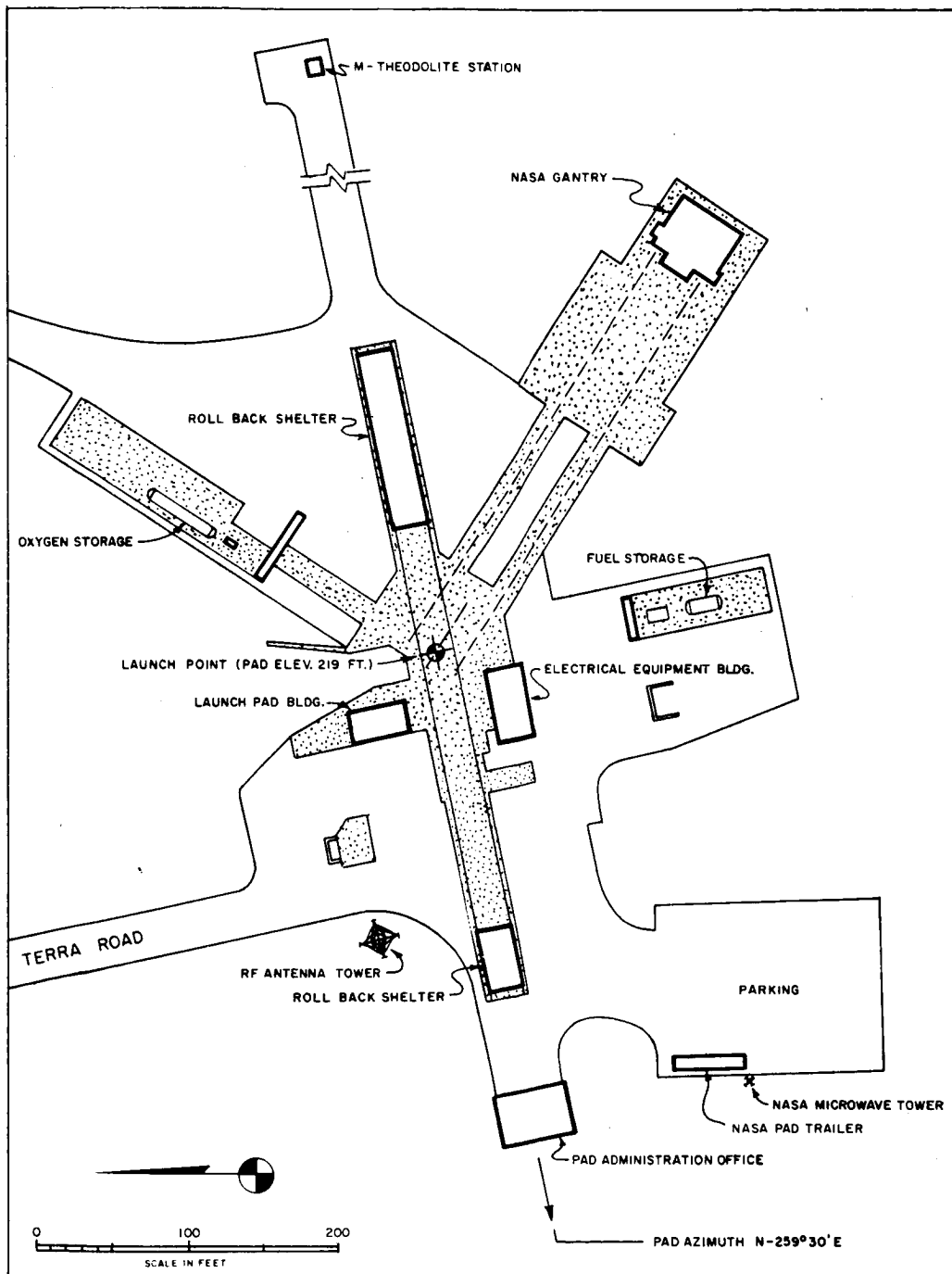


Figure 29. Pad 1, SLC-7

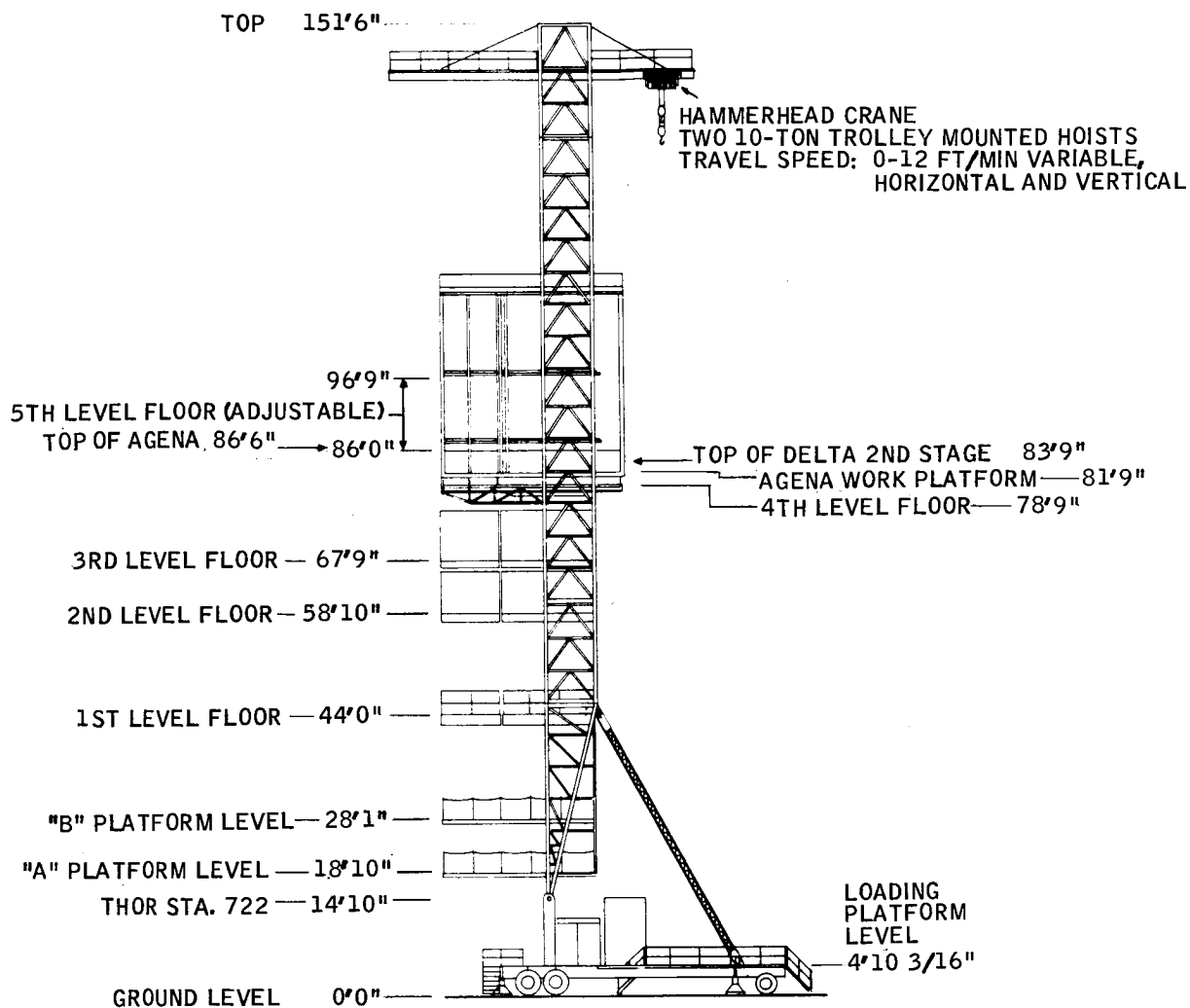


Figure 30. Side View of NASA Gantry

Spacecraft

The spacecraft design concept exerts a major influence upon the general support requirements. The significant design features of the HDS spacecraft are contained in the following description of the six major subsystems.

Structure. -- The spacecraft basic structure is a hexagonal cylinder domed at both top and bottom and is approximately 47 inches deep and 54 inches across its widest dimension as shown in Figure 31. Solar array panels, 44 inches long and 26 inches wide extend from each of the six sides of the spacecraft. These panels are hinged at their juncture with the spacecraft skin and remain folded during launch. The spacecraft is constructed in essentially two sections - a rigid and thermally conductive baseplate to which the experiment package is mounted and a "top hat" upper section containing a bulkhead on which the subsystem components are mounted. The baseplate section is thermally insulated from the upper section by an insulation blanket, and the two sections are connected through an insulating interface ring.

Power. -- Spacecraft power is derived from panel mounted solar cells which supply approximately 70 watts of continuous electrical power, supplemented at peak load conditions by a rechargeable battery to provide a total of 96 watts. Two nondissipative type regulators supply 28 and 5 volts for subsystem requirements.

Data handling. -- Data from the experiment and subsystems status instrumentation is collected, digitized, and stored in digital form. Upon command from a ground station, the data is read out and modulates a vhf carrier telemetry link to the station.

Communications. -- The communications subsystem, which operates in conjunction with the data handling subsystem, consists of a vhf command receiver, a vhf telemetry transmitter, and an S-band range and range-rate transponder. The communication frequencies and format are designed to be compatible with the NASA STADAN data acquisition network.

Attitude control. -- Spacecraft attitude deviations are sensed by a V-head earth horizon sensor and transmitted through the data handling system to the ground station via telemetry link. Return commands from the ground station activate magnetic coils to reorient the spacecraft. In a similar manner the spacecraft spin rate is maintained.

Experiment package. -- The experiment package consists of a radiometer with single 16-inch optics and having redundant detectors, calibration sources, power supplies, and all associated electronics. The detectors are maintained at cryogenic temperature by a dual cryogen sublimation refrigerator. Scanning of the earth's horizon is passively achieved through the spin rate of the spacecraft. The attitude and position of the spacecraft with respect to space is measured with redundant starmappers and sun sensors.

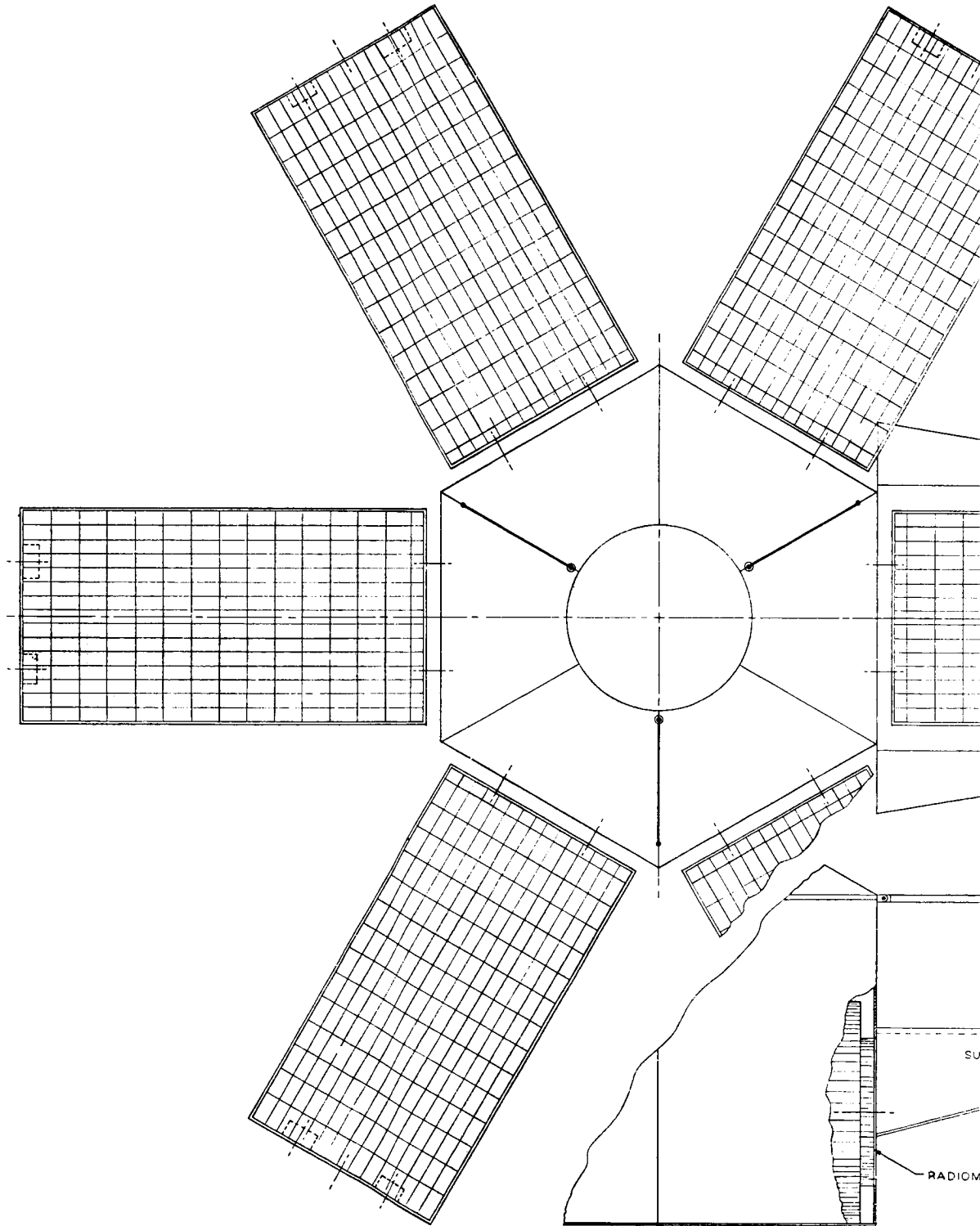
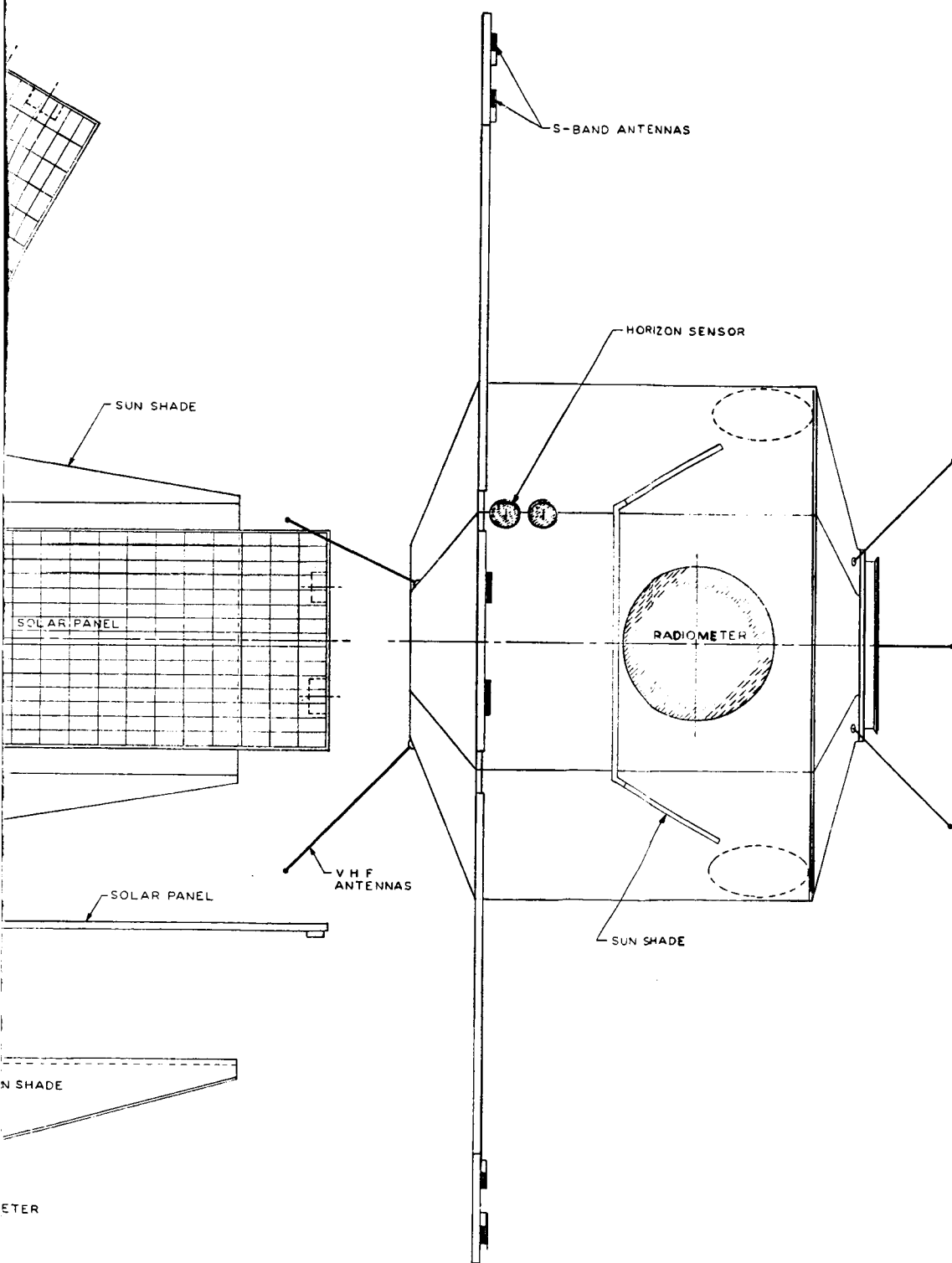


Figure 31. Conceptual Space



Spacecraft - External Layout

SUPPORT PLAN

The feasibility of the HDS mission is premised on a highly reliable spacecraft to achieve the one year lifetime objective. This objective must ultimately be translated into a test program designed to provide a high degree of confidence that the spacecraft will satisfactorily perform the mission.

Detailed test planning for the HDS program is a future development item. However, to give early visibility to the program support requirements, some preliminary test definition is necessary, and for this purpose the following statement of test philosophy and a test sequence are presented.

Test Philosophy

The long-life objectives of the HDS mission emphasize the requirements for spacecraft system reliability. The reliability requirements will be achieved through use of flight-proven components in systems designed with reliability as a primary goal. Verification of the spacecraft design will be accomplished during the development effort through a rigorous test and evaluation program. Maximum confidence in mission success will be assured by subjecting the complete spacecraft system to environmental exposures commensurate with the planned mission.

The test concept includes design qualification testing at component/subsystem and full spacecraft levels, followed by flight acceptance testing of each of three flight articles. Under this concept, launch site tests of the spacecraft will be limited to verification that all systems are functioning properly prior to launch and will consist principally of go, no-go checks on the launch pad.

Test Sequence

Development and evaluation of the spacecraft will logically follow a sequence similar to that diagrammed in Figures 32 and 33. The launch site functions are identifiable as a part of the test sequence through which the spacecraft passes from factory assembly to launch. A functional analysis was conducted of the identified launch site functions to define the support facility and equipment requirements.

Functional Analysis

To serve as an aid in identifying mission support requirements, a functional analysis of the prelaunch operations was performed. The test sequence presented previously established the primary launch site functions which were reduced to the next level subfunctions as shown in Table 6. The following description of spacecraft preparations is presented to provide continuity between the operations conducted at the launch site and the preceding test phase.

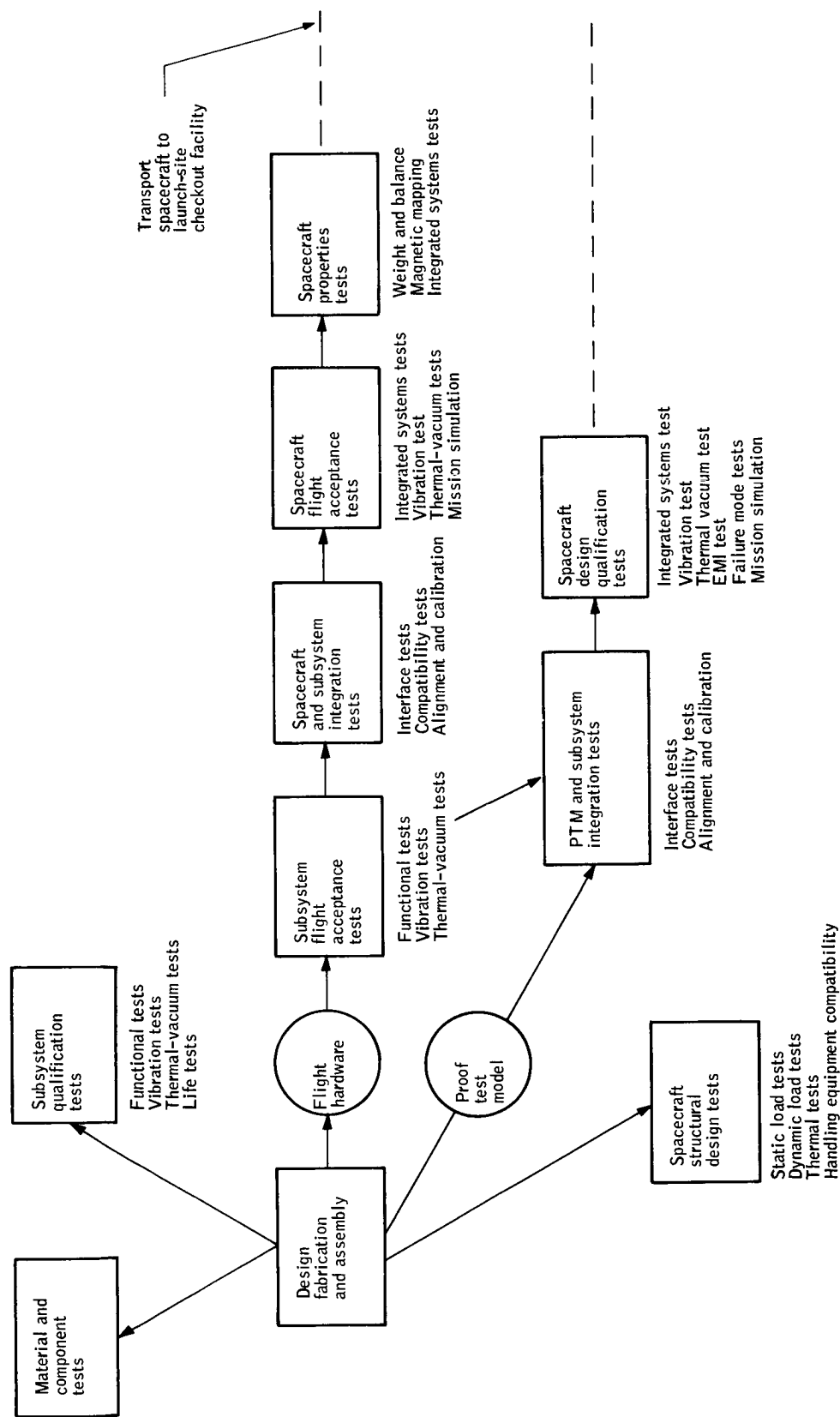


Figure 32. Test Sequence and Launch Site Functions - Factory Operations

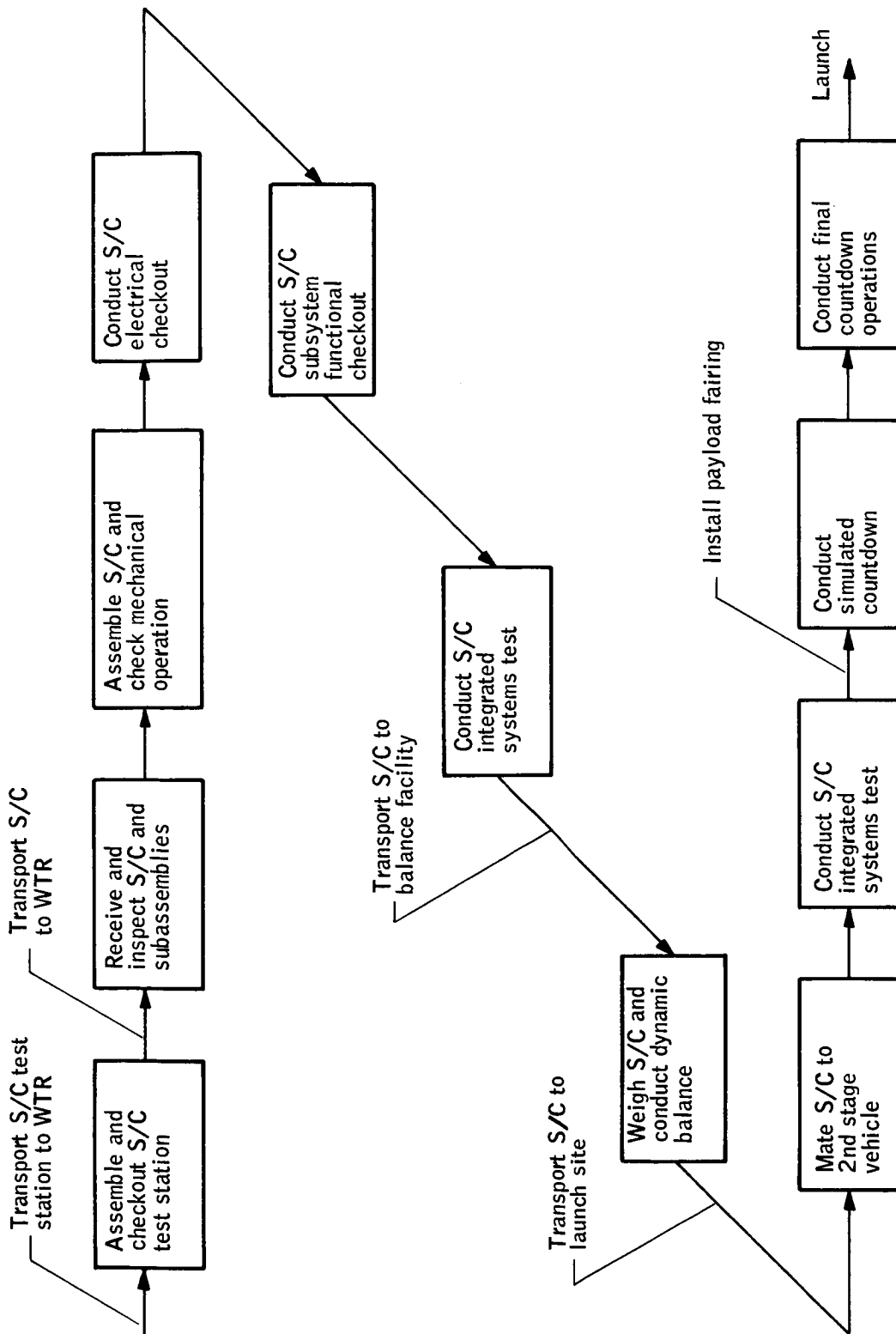


Figure 33. Spacecraft Launch Site Support Functions

TABLE 6. - LAUNCH SUPPORT FUNCTIONS AND REQUIREMENTS

Primary function	1 Assemble and check spacecraft test station	2 Receive and inspect spacecraft and subassemblies	3 Assemble spacecraft and check mechanical operation	4 Conduct spacecraft electrical check	5 Conduct spacecraft subsystem functional check
Objective	Assemble and check S/C test station at WTR assembly facility	Unload, inventory, and check condition of spacecraft following transport	Install items removed for transport and prepare for subsystem checks	Perform electrical isolation, continuity, and power checks	Verify functional performance of individual subsystems
Spacecraft configuration	Spacecraft less: 1) Solar panels 2) Batteries 3) Ordnance 4) Antennas	Spacecraft less: 1) Solar panels 2) Batteries 3) Ordnance 4) Antennas	Spacecraft less: 1) Solar panels 2) Batteries 3) Ordnance 4) Antennas	Spacecraft less: 1) Solar panels 2) Ordnance	Spacecraft less: 1) Solar panels 2) Ordnance
Facilities required	1) Spacecraft C/O facility 2) Environmental control	1) Spacecraft C/O facility 2) Environmental control	1) Spacecraft C/O facility 2) Environmental control	1) Spacecraft C/O facility 2) Environmental control	1) Spacecraft C/O facility 2) Environmental control
Subfunctions	1) Receive, inventory, and inspect test station components 2) Receive, inventory, and inspect S/C spare parts 3) Assemble S/C test station 4) Check S/C test station 5) Assemble S/C transport and handling equipment	1) Unload S/C container from carrier 2) Load container on transporter 3) Move container to C/O facility 4) Unload, remove S/C from container and mount on support fixture 5) Inspect S/C 6) Inventory parts	1) Install antennas 2) Install test batteries 3) Check solar panel deployment mechanism 4) Purge and pre-cool cryogenic cooler 5) Load and solidify cryogens 6) Maintain cryogenic cooler by circulating liquid helium	1) Hook up all electrical harnesses 2) Check all connectors for pin orientation 3) Connect external power 4) Verify electrical integrity of installation	1) Check all sensors 2) Calibrate T/M 3) Check rf systems 4) Operate all subsystems and verify performance
Support equipment requirements	1) S/C handling sling 2) Handling dolly 3) Environmentally controlled spacecraft container 4) S/C transporter 5) S/C support fixture 6) Inspection equipment	1) S/C support fixture 2) Cryogen supply and transfer system 3) Vacuum vent system to cooler 4) Liquid helium supply and circulating system	1) S/C support fixture 2) External electrical power 3) Electrical test set 4) Liquid helium supply and circulating system 5) Vacuum vent to cooler	1) S/C support fixture 2) External electrical power 3) Subsystem test equipment 4) Solar sensor simulator 5) Horizon sensor simulator 6) Test station to receive, decode and display T/M 7) Liquid helium supply and circulating system 8) Vacuum vent to cooler	

TABLE 6. - LAUNCH SUPPORT FUNCTIONS AND REQUIREMENTS - Concluded

Primary function	6 Conduct S/C integrated systems test	7 Prepare S/C and conduct final dynamic balance	8 Transport S/C to launch pad and mount on launch vehicle	9 Conduct S/C integrated systems test	10 Install payload fairing and conduct simulated countdown	11 Conduct final countdown operations
Objective	End-to-end tests of total S/C system	Determine mass characteristics of S/C and balance about proper axes	Mate and align S/C to second-stage vehicle and verify interface	Verify compatibility of S/C with range facilities and with launch vehicle	Install S/C in protective fairing and prepare for launch	Verify final readiness for launch
Spacecraft configuration	Spacecraft less: 1) Solar panels 2) Ordnance	Spacecraft less 1) Solar panels 2) Ordnance	Spacecraft in injection configuration less ordnance	Spacecraft in injection configuration less ordnance	Spacecraft in injection configuration	Spacecraft in fairing
Facilities required	1) Spacecraft C/O facility 2) Environmental control	1) Explosive safe area - spin facility 2) Environmental control	1) Launch pad 2) Environmental control 3) Umbilical mast	1) Launch pad 2) Environmental control 3) Umbilical mast	1) Launch pad 2) Environmental control 3) Umbilical mast	1) Launch pad 2) Environmental control 3) Umbilical mast
Subfunctions	1) Supply test stimuli to all S/C sensors and command S/C into its operating modes 2) Monitor all S/C in-orbit parameters plus special check-out parameters	1) Conduct electrical check of solar arrays 2) Install solar panels and flight batteries 3) Supercool cryogenic cooler 4) Install S/C in handling container 5) Transport S/C to spin facility 6) Remove S/C, install ordnance, and mount balance fixture 7) Balance S/C 8) Remove S/C ordnance 9) Install S/C in handling container	1) Transport S/C container to launch pad 2) Erect S/C container on launch vehicle 3) Install protective curtain around S/C container 4) Remove handling container 5) Check mechanical alignment of S/C to launch vehicle 6) Connect and verify umbilical interface to cryogenic cooler	1) Supply test stimuli to S/C sensors and command S/C into its operating modes 2) Monitor all S/C in-orbit parameters plus special check-out parameters 3) Check S/C rf systems and for stray voltage 4) Install S/C ordnance	1) Install and check alignment of S/C fairing 2) Conduct simulated countdown with S/C on battery power 3) Clean optics 4) Purge fairing with dry, inert atmosphere and maintain at slight positive pressure 5) Monitor S/C go/no-go parameters	1) Conduct countdown sequence 2) Monitor S/C go/no-go parameters
Support equipment requirements	1) S/C support fixture 2) External electrical power 3) System test stimuli 4) Command transmitter 5) Beacon receiver 6) Test station to receive, decode, and display T/M 7) Liquid helium supply and circulating system 8) Vacuum vent to cooler	1) S/C support fixture 2) Solar array illuminator 3) Environmentally controlled handling container 4) Transporter 5) S/C handling sling 6) Handling dolly 7) S/C spin fixture adapter 8) Liquid helium supply and circulating system 9) Vacuum vent to cooler	1) Environmentally controlled handling container 2) Transporter 3) S/C handling sling 4) External electrical power 5) Environmentally controlled protective cover for S/C 6) Vacuum vent to cooler	1) External electrical power 2) System test stimuli 3) Command transmitter 4) Beacon receiver 5) Test station to receive, decode, and display T/M 6) Environmentally controlled protective cover for S/C 7) Vacuum vent to cooler	1) External electrical power 2) Vacuum vent to cooler 3) Supply of dry inert gas and circulating system 4) Test station to receive, decode, and display T/M	1) External electrical power 2) Vacuum vent to cooler 3) Supply of dry inert gas and circulating system 4) Test station to receive, decode, and display T/M

As a part of flight acceptance tests, a final spacecraft systems check will be conducted, the research package alignment verified, and the spacecraft balanced. In preparation for shipment, the cryogenic cooler will be emptied and purged and the solar panels, batteries, and antennas removed. The spacecraft will then be placed in a sealed, transportable container provided with an inert, dry atmosphere for shipment.

Upon arrival at the launch site, the operations depicted in Table 6 begin. The spacecraft support equipment requirements for each operation are identified in the final rows of the table and may logically be categorized into the following groups:

- Transporting, assembly, and handling - This group includes the equipment required to protect, transport, lift, and position the spacecraft during assembly and test.
- Subsystem and system test - This group includes the equipment required for checkout and testing of the spacecraft.
- Servicing and maintenance - This group includes the equipment required to service and maintain the spacecraft subsystems during test and check-out.

Design specification of the equipment identified in each of these groups is an item for future phase planning. At this time the design of transport, assembly, and handling equipment appears to offer no significant problems. However, the servicing and maintenance requirements of the HDS spacecraft system, as well as the test and check-out concept, are of interest and are further discussed below.

The support equipment requirements identified in the table indicate a potential problem area in the HDS mission support concept, resulting from the support needs of the cryogenic cooler which is an integral part of the spacecraft research package.

The cryogenic cooler proposed for the HDS mission is shown in the cutaway view of Figure 34. A basic feature of the cooler is that it uses two different solid cryogenics: the inner chamber contains neon which acts as the detector coolant; the outer shell contains methane which shields the inner container from heat penetrating the system insulation. An evaporation vent, which also serves as a filler port, is provided for each cryogen. During operation in space, the solidified cryogenics are slowly sublimed by absorbing heat from the detector and the surroundings.

The cryogenics may be loaded into the container in gaseous form and solidified by circulating liquid helium through the heat exchanging coils shown in the diagram. Maintenance of the cooler under earth atmosphere conditions (with some depletion of the coolant) may be accommodated for short time periods (< 7 days) by supercooling the cryogenics. Normal maintenance for longer periods will require a continuous circulation of the liquid helium coolant supply.

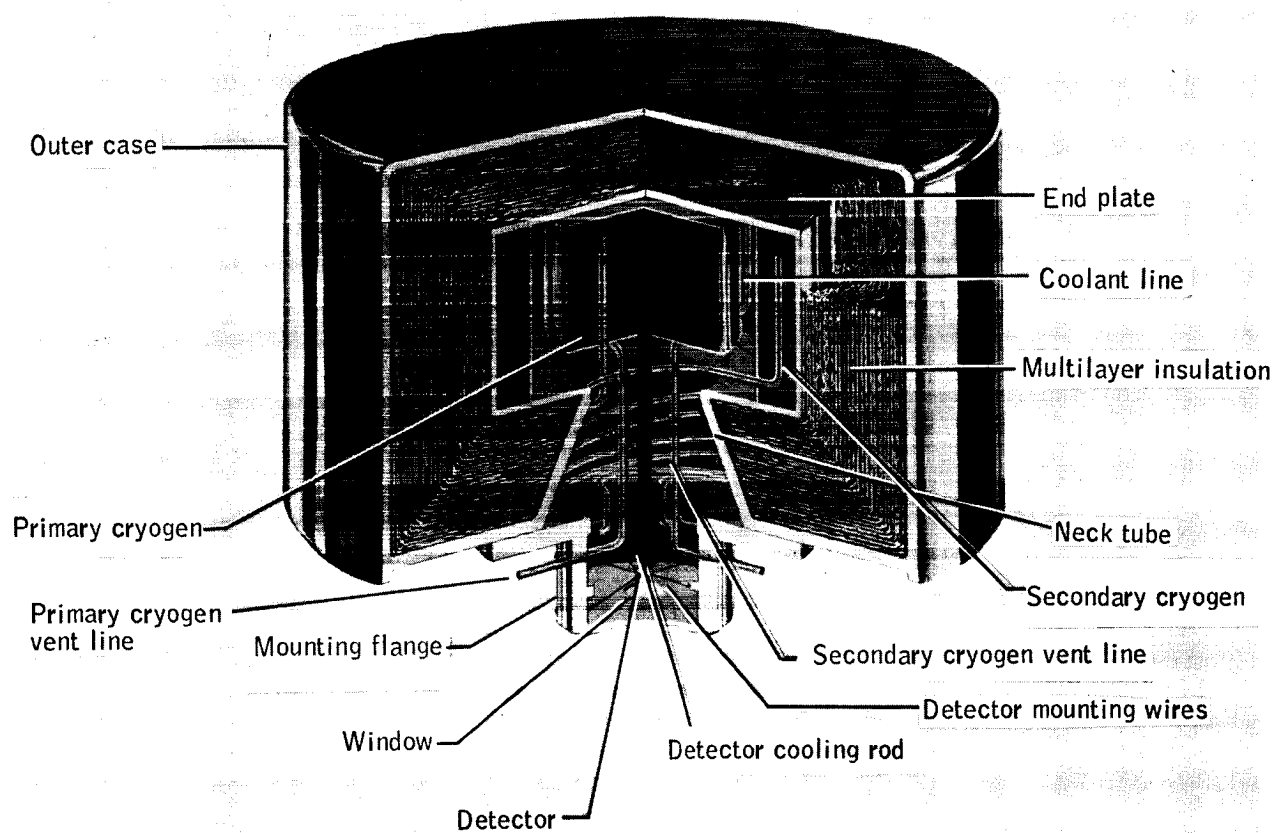


Figure 34. Conceptual View of Two-Cryogen Refrigerator

The shielding cryogen, methane, produces a potentially hazardous vapor which must be continually vented to a safe area during ground operations.

Prior to launch, the spacecraft and fairing must be purged to rid them of any accumulated gases which might subsequently influence the radiance measurements of the spacecraft. Gaseous nitrogen or argon would be acceptable as purgants, since neither influences the carbon dioxide band which the radiometer is designed to measure.

Test and Check-out Concept

The current study phase of the HDS Spacecraft provides a level of design detail which precludes a detailed definition of test requirements. However, within the general mission constraints discussed previously, a preliminary test and check-out concept may be outlined.

Spacecraft testing at the launch site will be primarily directed at the system level to verify that the spacecraft is functioning properly prior to launch. Final check-out from the launch pad will be by rf link to the spacecraft assembly building. The spacecraft test station at the launch site must provide essentially parallel communications capability with that of the NASA STADAN data acquisition stations. That is, it must provide the capability to command the spacecraft into its various modes of operation and monitor the system response via the telemetry link. It must also provide external stimuli for end-to-end testing, a central source for controlling and sequencing tests, and an external source of power to supplement the spacecraft solar panels.

A simplified block diagram illustrating the test station concept is presented in Figure 35.

Ground power and electrical test stimuli to the spacecraft will be provided through the second-stage umbilical. External simulators will be provided to simulate inputs to the optical sensors. Spacecraft responses to test stimuli will be directed through the on-board data handling subsystem and transmitted to the test station via the rf link. The in-flight calibrator for the radiometer and starmapper will provide the source of system stimuli for the experiment package verification.

Special Equipment Requirements

To support the HDS operational plan, the following special items of equipment will be required.

Support of cryogenic cooler. -- For charging and subsequent servicing of the cooler, it will be necessary to provide a supply of gaseous neon and gaseous methane. The containers will be pressurized to allow transfer of the gases to the spacecraft.

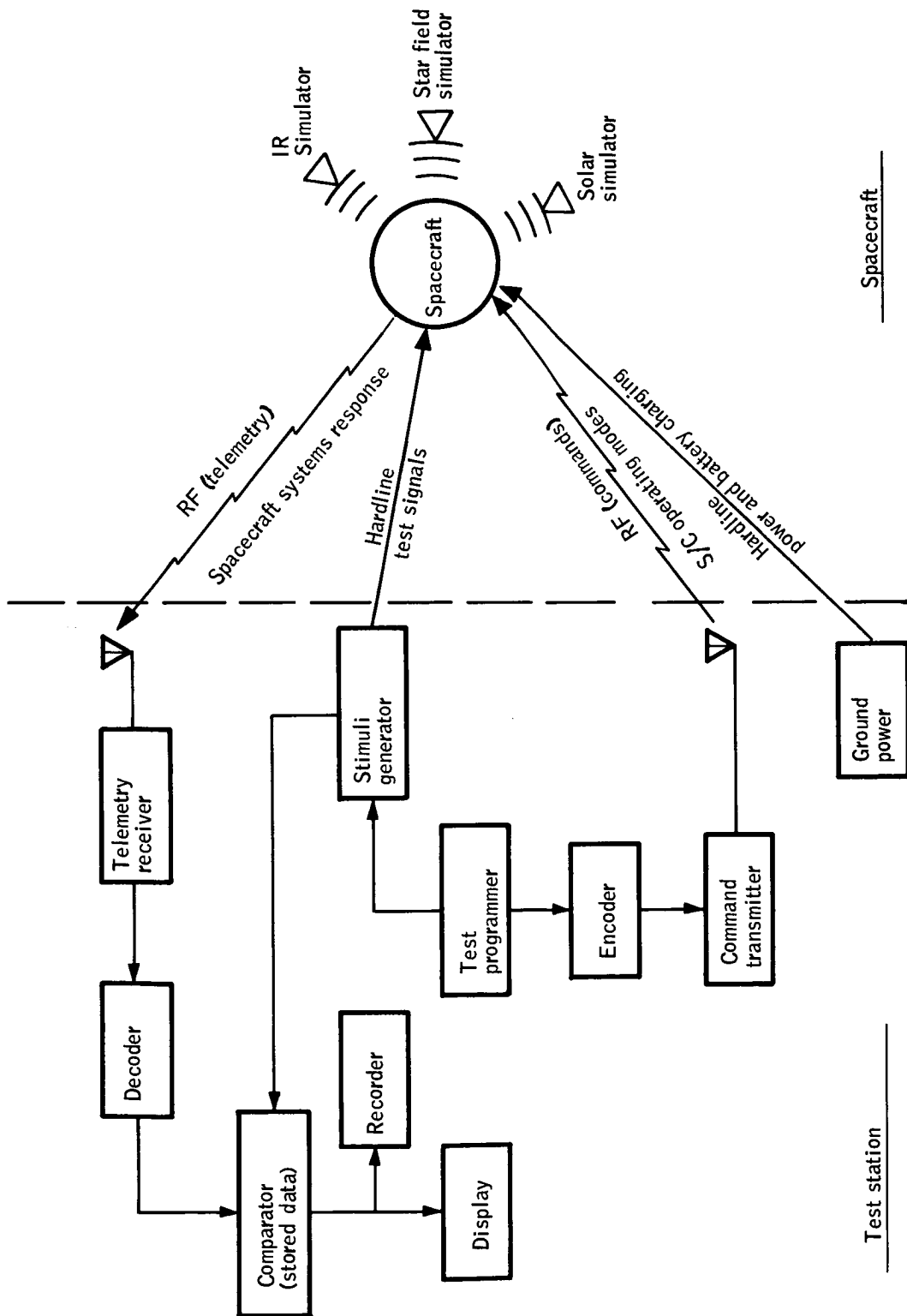


Figure 35. Check-Out System Block Diagram

The initial servicing requirements of the containers are small, requiring approximately 2 liters of neon and 2.5 liters of methane. Solidifying of the cryogens will require a supply of liquid helium and a method for circulating it through the heat exchanging coils which are a part of the spacecraft cryogenic container. Maintenance of the cooler prior to launch will require a vacuum vent system capable of providing an industrial-level vacuum.

In general, support requirements of the cooler appear to pose no unique design problems and can be met without significant modification to existing WTR launch site facilities. However, when these requirements are defined, they should be coordinated with WTR as early as possible in the program to allow maximum lead time and efficiency for incorporation of any required modifications.

Test station. - A spacecraft test station will be needed to support the testing necessary to verify system performance. The design requirements of the test station are dependent upon final spacecraft systems definition and for now can only be identified in a general sense. The test station must provide the capability to:

- Command the spacecraft via rf link/or hardline
- Receive, decode, and display spacecraft response parameters via rf telemetry link
- Provide system stimuli suitable for end-to-end test of the spacecraft systems
- Provide and monitor electrical power to supplement the spacecraft solar array source.

The prelaunch test requirements of the HDS spacecraft are common to those of previous spacecraft programs, and no unique test station design problems are foreseen.

CONCLUSIONS AND RECOMMENDATIONS

The following conclusions have been drawn from the flight vehicle operations study:

- An Improved Delta launch vehicle, Model DSV-3N, is compatible with the HDS spacecraft design concept and mission requirements and will be available during the operational time period of the program.
- Significant increases in the probability of mission success may be achieved by employing multiple flights in the operational plan.

- The launch operation facilities and service and the support program approach of the Western Test Range are compatible with the basic support requirements of this program, with the exception of the cryogenic cooler. This could involve the addition of coolant fill and vent lines on the launcher.
- Support requirements at the launch site are those typical for spacecraft check-out and vehicle interface with no unusual calibration or alignment equipment requirements.

It is recommended that consideration be given to a modification to the presently planned second-stage Delta vehicle spin table to accommodate the low (3 rpm) spin rate required for the HDS spacecraft. The initial spin-up could be accomplished with a highly reliable electric or pneumatic motor without adding appreciably to the complexity of the spin table. This approach is recommended over a spin-up or despin system on the spacecraft because it keeps additional systems and possible failure areas away from the spacecraft. This spin-up requirement should be established early enough in the program to allow this capability to be incorporated into the currently planned Delta spin table with minimum costs to the HDS program.

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APPENDIX A
STATE DIAGRAMMING TECHNIQUE

APPENDIX A STATE DIAGRAMMING TECHNIQUE^a

INTRODUCTION

The advent of large-scale digital and analog computers provides a means for the analysis and design of highly complex systems. However, in order to use these computers efficiently, it is necessary to develop new approaches and concepts for the characterization and description of the system.

Within the past several years the state space approach has gained prominence in the analysis of large and complex systems, including some of the most advanced work on optimal systems. It is also possible to employ the same analytical concepts in the system reliability calculations and thus obtain a common framework for the generalized system studies.

The approach is based on the classical Markov methods that have been known for a considerable time but have found only a limited application in reliability work. Using the state space approach considerably simplifies the analytical workload by employing computer methods in their full capacity. It is also possible to replace the cumbersome Boolean methods and reliability block diagrams with the state transition diagrams and state equations.

In the description that follows, a generalized introductory treatment and examples of application are presented. For details concerning the Markov methods, standard texts (such as Feller or Loeve) on the theory of probability provide reference.

THE CONCEPT OF STATE SPACE

The possible conditions of a system are called "states", and their totality is called "state space" or "phase space". A state can often be defined by listing the equipments that are working satisfactorily. The states selected should be mutually exclusive and exhaustive so that at any instant in time a system (a given equipment configuration) exists only in one of a number of possible states.

As an example, consider a simple redundant system consisting of three parallel channels. The states in this system will be:

1. Three channels operating satisfactorily
2. Two channels operating satisfactorily

^aPukite, J.: Development of A GENERALIZED RELIABILITY MODEL.
R-ED-1582, Aeronautical Division Minneapolis, Honeywell Inc.,
13 September 1965.

3. One channel operating satisfactorily
4. No operating channels (system failure)

The state space in this case consists of the four states listed above.

STATE DIAGRAM

After the state space has been defined, the next task is to construct a mathematical model. A graphical model will be used in preliminary analysis. Later, however, as the system analyst gains more experience this graphical presentation can be replaced by the state equations. Following the conventional notation, we let E_0, \dots, E_N denote the states of the system. We will use an arrow to indicate the direction of the possible transition from state to state.

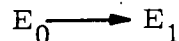
In order to gain familiarity with the state diagrams we will consider some practical examples.

Single-Element State Diagram

Let the states be designated as follows:

- E_0 - element working satisfactorily
- E_1 - element not working satisfactorily (failed state)

The state diagram in this case will be represented as:



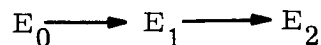
Note that a transition is possible only from state E_0 to E_1 .

Two-Element State Diagram (Identical Elements)

In this case with the elements operating in parallel the states can be designated as follows:

- E_0 - both elements working satisfactorily
- E_1 - one element working satisfactorily
- E_2 - neither element working satisfactorily (failed state)

The corresponding state diagram will be:



N-Element State Diagram (Identical Elements)

Considering the case where N elements are operating in parallel we obtain the following states:

- E_0 - N elements operating satisfactorily
- E_1 - (N-1) elements operating satisfactorily
-
- E_{N-1} - one element operating satisfactorily
- E_N - no operating elements

In graphical form:



STATE TRANSITION RATES

To extend our mathematical model we next consider the transition rates between the various states. In this introductory explanation we are concerned entirely with system following Poisson's process defined by the postulate:

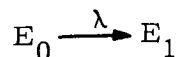
Whatever the number of changes during the time interval $(0, t)$, the probability that a change occurs during $(t, t+h)$ is $\lambda h + o(h)$, and the probability that more than one change occurs is $o(h)$.

In the above postulate, λ is the transition rate, which in reliability work is called failure rate, and $o(h)$ is used as a general designation for terms involving second and higher order power terms in h .

Failure rates (transition rates) will be indicated on the state diagram by placing them above the corresponding arrows. We can now complete the state diagrams that were previously started.

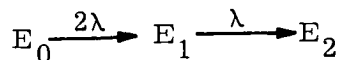
Single-Element State Diagram

Letting the element failure rate equal λ , the state diagram for this system is:



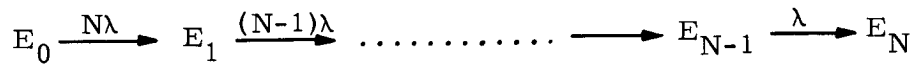
Two-Element Parallel System

For this system the state diagram is (all elements identical):



N-Element Parallel System

The state diagram for this system is:

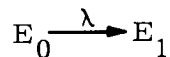


STATE EQUATIONS

Once the state diagrams showing the transition rates have been derived, the next step is to derive the system state equations. We let $P_0(t)$ $P_N(t)$ denote the probability that the system is in state E_0 E_N at time t and make use of the Poisson process postulate stated in the previous section. The following examples will illustrate the method used in obtaining the state equations.

Single-Element System

The state diagram for this configuration was derived earlier:



The probability $P_0(t+h)$ that the element is in state E_0 at time $t+h$ is the probability that it was in state E_0 at time t and that no change took place in the time interval $(t, t+h)$. Thus:

$$P_0(t+h) = P_0(t) (1 - \lambda h + o(h))$$

Rewriting the above equation, we have:

$$\frac{P_0(t+h) - P_0(t)}{h} = - \frac{\lambda h P_0(t)}{h} + \frac{o(h)}{h}$$

Taking the limit as $h \rightarrow 0$, we obtain:

$$\lim_{h \rightarrow 0} \frac{P_0(t+h) - P_0(t)}{h} = \frac{dP_0(t)}{dt} = P'_0(t)$$

and

$$\lim_{h \rightarrow 0} \frac{o(h)}{h} = 0$$

Then:

$$P'_0(t) = -\lambda P_0(t)$$

Similarly for state E_1 , the probability that the element is in this state at time $t+h$ is the sum of the following probabilities:

1. The probability that the element is in state E_1 at time t
2. The probability that the element is in state E_0 at time t and a change takes place during the interval $(t, t+h)$

Thus:

$$P_1(t+h) = P_1(t) + P_0(t) [\lambda h + o(h)]$$

Rewriting the equation as in the previous case and then taking the limit as $h \rightarrow 0$ we obtain:

$$P'_1(t) = \lambda P_0(t)$$

The two equations derived above together with the initial conditions are called the state equations of the system. In our case the state equations are:

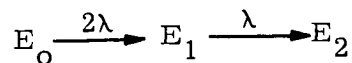
$$P'_0(t) = -\lambda P_0(t) \quad P_0(0) = 1$$

$$P'_1(t) = \lambda P_0(t) \quad P_1(0) = 0$$

Using a similar process we can derive the state equations for the other two examples considered earlier.

Two-Element Parallel System

The state diagram for this system is repeated for reference:



The state equations are derived as follows:

$$P_0(t+h) = P_0(t) (1 - 2\lambda h) + o(h)$$

$$P_1(t+h) = P_0(t) 2\lambda h + P_1(t) (1 - \lambda h) + o(h)$$

$$P_2(t+h) = P_1(t) \lambda h + P_2(t) + o(h)$$

In the limiting case, as $h \rightarrow 0$

$$P'_0(t) = -2\lambda P_0(t)$$

$$P'_1(t) = 2\lambda P_0(t) - \lambda P_1(t)$$

$$P'_2(t) = \lambda P_1(t)$$

The initial conditions are:

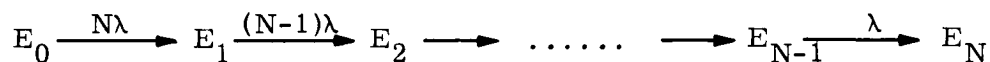
$$P_0(o) = 1$$

$$P_1(o) = 0$$

$$P_2(o) = 0$$

N-Element Parallel System

From the previously derived state diagram:



we can write the following equations:

$$P_0(t+h) = P_0(t) (1 - N\lambda h) + o(h)$$

$$P_1(t+h) = P_0(t) N\lambda h + P_1(t) [1 - (N-1)\lambda h] + o(h)$$

.....

$$P_N(t+h) = P_{N-1}(t) \lambda h + P_N(t) + o(h)$$

In the limiting case, as $h \rightarrow 0$, we have:

$$P'_0(t) = -N\lambda P_0(t)$$

$$P'_1(t) = N\lambda P_0(t) - (N-1)\lambda P_1(t)$$

.....

$$P'_N(t) = \lambda P_{N-1}(t)$$

The initial conditions are:

$$P_0(o) = 1, P_1(o) = 0, \dots, P_{N-1}(o) = 0, P_N(o) = 0$$

COMPUTATIONAL METHODS

The system state equations are nothing but a set of simultaneous linear differential equations. The computational methods for the solution of linear differential methods are well known and widely described in mathematical literature. The following three approaches should be considered and the most appropriate selected for the particular system under analysis:

1. Numerical solution using Laplace transform method
2. Analog computer solution
3. Digital computer solution

These three methods are briefly discussed in this section.

Laplace Transform Solution

As an illustration of the application of Laplace transform techniques the two-element parallel system will be considered. The state equations for this system as previously derived are:

$$P'_0(t) = -2\lambda P_0(t)$$

$$P'_1(t) = 2\lambda P_0(t) - \lambda P_1(t)$$

$$P'_2(t) = \lambda P_1(t)$$

with the following initial conditions:

$$P_0(o) = 1, P_1(o) = 0, P_2(o) = 0$$

Taking the Laplace transform of the above we obtain:

$$S P_0(S) - 1 = -2\lambda P_0(S)$$

$$S P_1(S) = 2\lambda P_0(S) - \lambda P_1(S)$$

$$S P_2(S) = \lambda P_1(S)$$

From the above, we derive:

$$P_0(S) = \frac{1}{S + 2\lambda}$$

$$P_1(S) = \frac{2\lambda P_0(S)}{S + \lambda} = \frac{2\lambda}{(S + \lambda)(S + 2\lambda)}$$

$$P_2(S) = \frac{\lambda P_1(S)}{S} = \frac{2\lambda^2}{S(S + \lambda)(S + 2\lambda)}$$

Taking the inverse Laplace transforms:

$$P_0(t) = e^{-2\lambda t}$$

$$P_1(t) = 2e^{-\lambda t} - 2e^{-2\lambda t}$$

$$P_2(t) = 1 - 2e^{-\lambda t} + e^{-2\lambda t}$$

We recognize the last expression as the well-known equation for the probability of failure for a two-element parallel system. We also note that:

$$P_0(t) + P_1(t) + P_2(t) = 1$$

for any arbitrary t .

Other system state equations can be handled using the same approach.

Analog Computer Solution

The analog computer is particularly well suited for solving linear differential equations. The state equations can be set up directly and require only amplifiers, integrators, and coefficient potentiometers. Initial conditions can be introduced in the conventional manner by applying the properly scaled voltages

to the corresponding integrator initial condition terminal. State probabilities can be plotted out directly on the X-Y plotter as functions of time. The analog computer approach appears to be especially useful in system tradeoff studies because a large number of tradeoff curves can be obtained with minimum expenditure of time. The accuracy problem is probably the only disadvantage.

Digital Computer Solution

Where a general-purpose digital computer is available, the system state equations can be conveniently solved by means of the widely available subroutines for handling linear differential equations. Another approach would be to make use of the analog computer simulation program on the digital computer.

In the case where differential equation subroutines are used, the state equations should preferably be expressed in matrix form as follows:

$$[P'(t)] = [A] [P(t)]$$

where $[A]$ is the state transition matrix. As an example let us consider the two-element parallel system. The state equations for this system can be written in matrix form as follows:

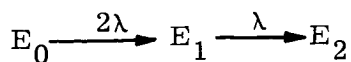
$$\begin{bmatrix} P'_0(t) \\ P'_1(t) \\ P'_2(t) \end{bmatrix} = \begin{bmatrix} -2\lambda & 0 & 0 \\ 2\lambda & -\lambda & 0 \\ 0 & \lambda & 0 \end{bmatrix} \begin{bmatrix} P_0(t) \\ P_1(t) \\ P_2(t) \end{bmatrix}$$

APPLICATIONS

State diagrams can be used to evaluate and rate the various redundancy configurations. They can be also very useful in other reliability calculations involving multiple failure modes, maintainability problems, availability problems and as an input to other more complex reliability models, such as system effectiveness and figure-of-merit models. Examples of these applications are given below.

Two-Element Redundant Systems (Identical Elements)

Parallel redundancy. -- This type of configuration was previously considered. Its state diagram is:



Standby redundancy. -- This type of redundancy yields the following state diagram:

$$E_0 \xrightarrow{\lambda} E_1 \xrightarrow{\lambda} E_2$$

Since the transition rate from E_0 to E_1 in the standby system is lower than in the parallel arrangement, standby redundancy will yield a higher reliability.

Three-Element Redundant Systems (Identical Elements)

Parallel redundancy. -- The state diagram is:

$$E_0 \xrightarrow{3\lambda} E_1 \xrightarrow{2\lambda} E_2 \xrightarrow{\lambda} E_3$$

Standby redundancy. -- The state diagram is:

$$E_0 \xrightarrow{\lambda} E_1 \xrightarrow{\lambda} E_2 \xrightarrow{\lambda} E_3$$

Two out of three (majority voting). -- The state diagram is:

$$E_0 \xrightarrow{3\lambda} E_1 \xrightarrow{2\lambda} E_2$$

Pair and a spare redundancy. -- The state diagram is

$$E_0 \xrightarrow{2\lambda} E_1 \xrightarrow{\lambda} E_2$$

Majority voting with switching. -- The state diagram is:

$$E_0 \xrightarrow{3\lambda} E_1 \xrightarrow{\lambda} E_2$$

We note that the three element configurations yield either three or four states. In the case of identical elements it is obvious that the redundancy scheme resulting in more states will yield higher overall reliability since the average time required to reach the failed state will be longer.

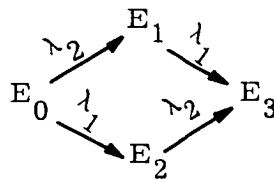
Redundant Systems with Different Elements

Up to this time we considered only those redundant configurations with identical elements. In the case of nonidentical elements our state diagram will have to be modified. This modification is relatively easy and an example will be used to illustrate the method.

Let us consider a two-element parallel system with element failure rates λ_1 and λ_2 . The states will be assigned as follows:

- E_0 - both elements operating
- E_1 - element 1 operating
- E_2 - element 2 operating
- E_3 - no operating elements (failed state)

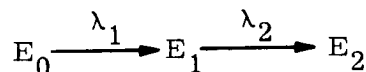
The system state diagram in this case can be drawn as follows:



The diagram is simpler in the case of a two-element standby system with unequal failure rates. Its states are:

- E_0 - primary element operating
- E_1 - secondary element operating
- E_2 - failed state

The state diagram follows immediately:



Redundant System with Switching

The state diagrams for a redundant system involving failure detection and switching can also be easily obtained using the same techniques as illustrated in the following example:

Consider a standby system where the primary channel contains an attached failure-sensing and switching device. Let us further assume that the switching device can fail only in one of two modes: failure of the type where the switching function is suppressed (Type I) and failure resulting in a false switching to the secondary channel (Type II). The failure rates are as follows:

- λ - primary and secondary element failure rate
- λ_{S1} - switching element failure rate (Type I failure)
- λ_{S2} - switching element failure rate (Type II failure)

Switching element failure rates are assigned in such a way that:

$$\lambda_{S1} + \lambda_{S2} = \lambda_S$$

The system states are defined as:

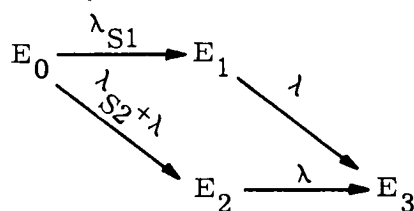
E_0 - primary element and switching element operating satisfactorily

E_1 - primary element operating satisfactorily, switching element unable to switch (Type I) failure)

E_2 - secondary element operating satisfactorily

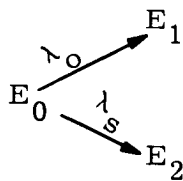
E_3 - failed state

The state diagram in this case is:



Multiple Failure Modes

Multiple failure modes in a single element can also be handled without difficulty. In our model we will consider the failed states to be permanent, i. e., there will be no transitions between the failed states. The simplest example is that of only two different failure modes, such as open and short in an electronic device. Let the corresponding failure rates be denoted by λ_o and λ_s . Then it follows that the following state diagram applies:



where:

E_0 - operating state

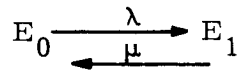
E_1 - open failure

E_2 - short failure

This type of model will be particularly useful in failure mode and failsafety analysis.

System with Repairs

In our previous discussion we considered only one-directional transitions, i.e., all the failed states were permanent. It is possible, however, without much difficulty to extend the state space model to systems where repair action is permissible. Since this area is well documented in maintainability and availability analyses, only a simple example is given here. Assume that we have a simple system consisting of a single element with a failure rate λ and repair rate μ . In this case there are only two possible states: E_0 , system operating, and E_1 , system not operating. The state diagram in this case is very simple.



The above model can be easily extended to more complex cases by following the general approach outlined previously.

Other Applications

Many workers in the reliability field have indicated that reliability alone is not a very good measure of a system's worth and have proposed various schemes considering the effects of the intermediate states. The best known of these are probably the system effectiveness and reliability figure-of-merit models. Since our state space model yields directly all the different state probabilities, this model does not require any modifications and can be used directly in these schemes.

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APPENDIX B
LAUNCH OPERATION STUDIES SUPPLEMENTARY FIGURES

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APPENDIX B

LAUNCH OPERATION STUDIES SUPPLEMENTARY FIGURES

MISSION OPERATIONAL RELIABILITY

The following figures illustrate the variation in mission operational reliability with spacecraft failure rate for three assumed booster reliabilities and various numbers of reserve vehicles.

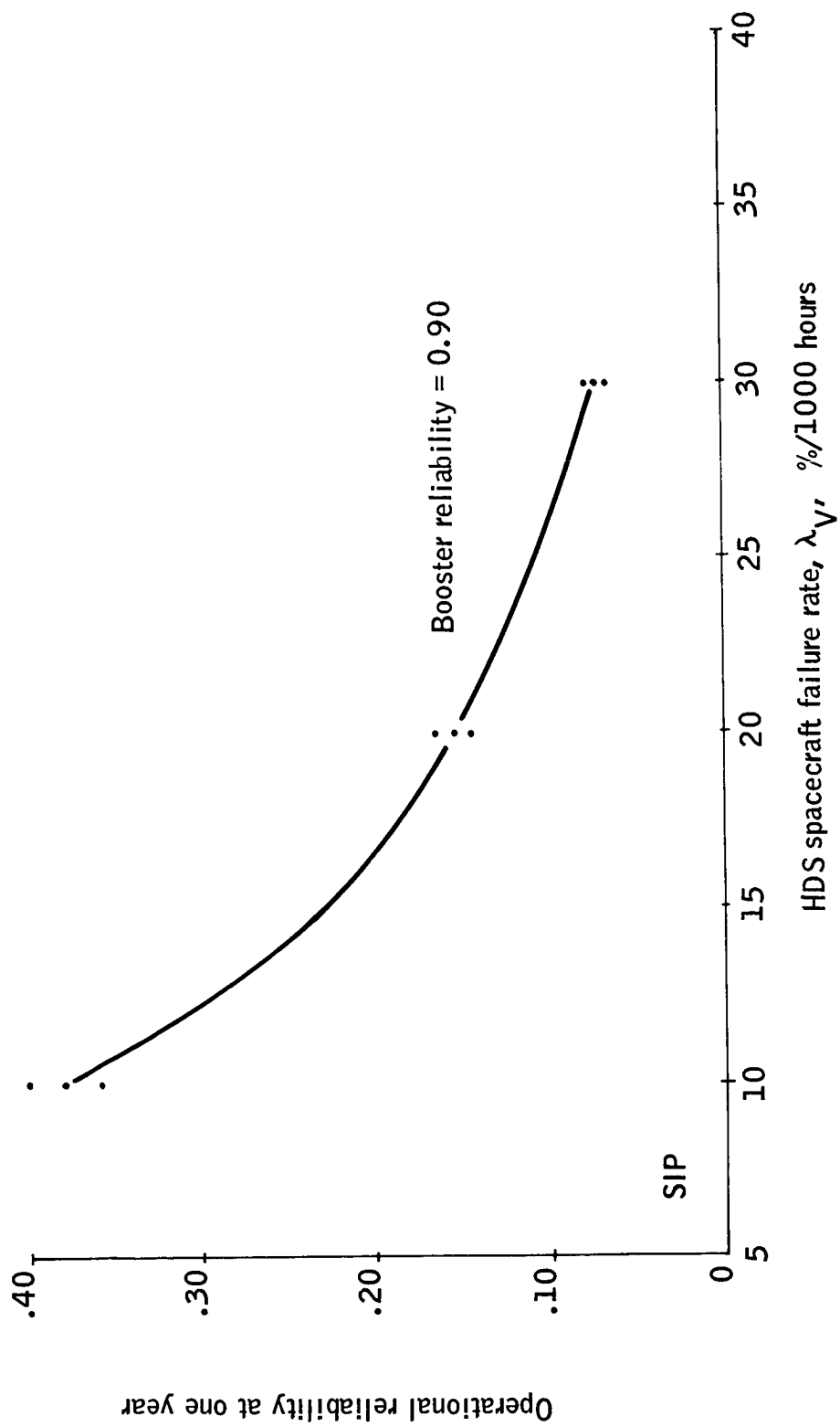


Figure B1. Mission Operational Reliability, No Reserve Program

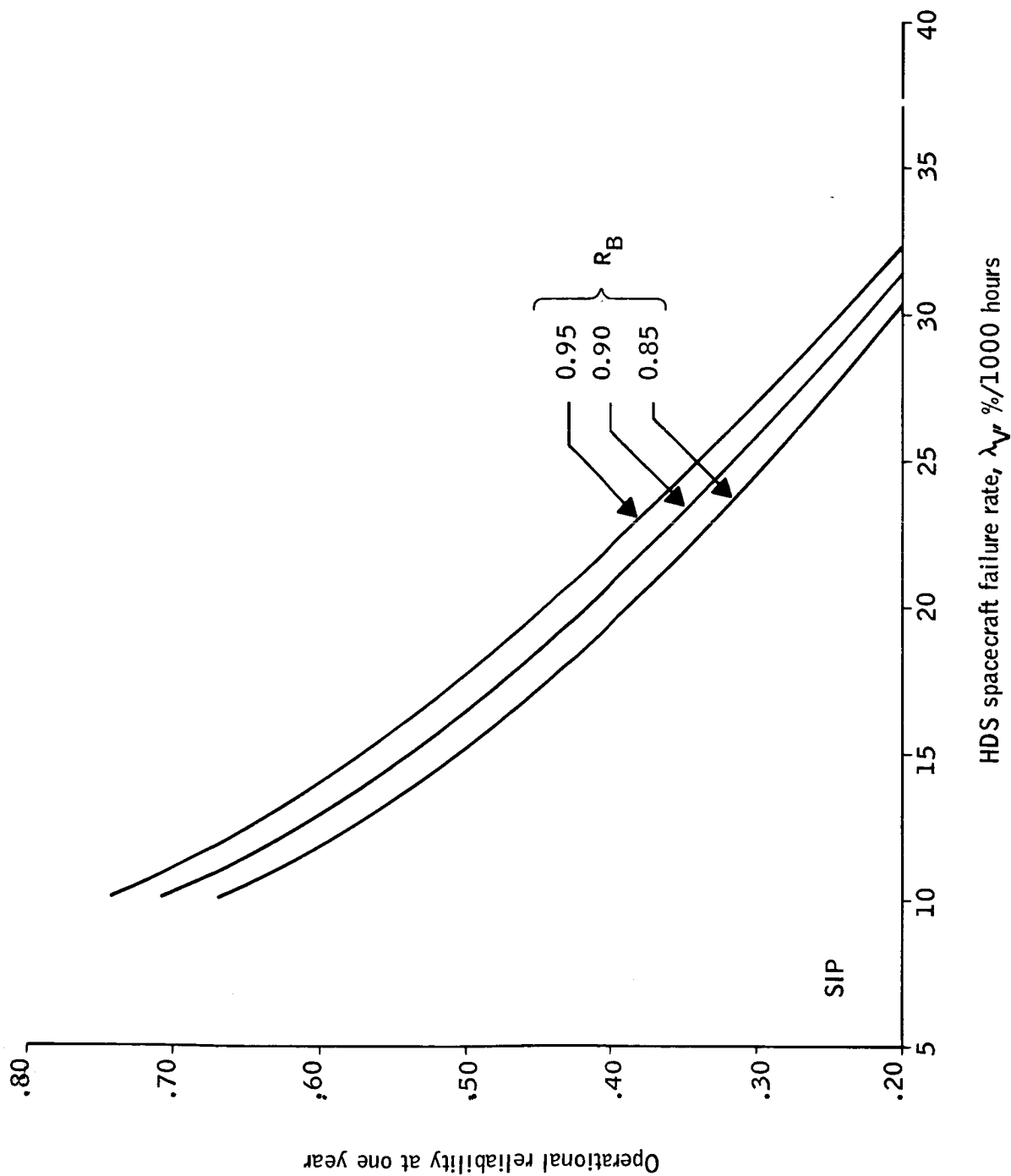


Figure B2. Mission Operational Reliability, One Reserve Unit Program Utilization

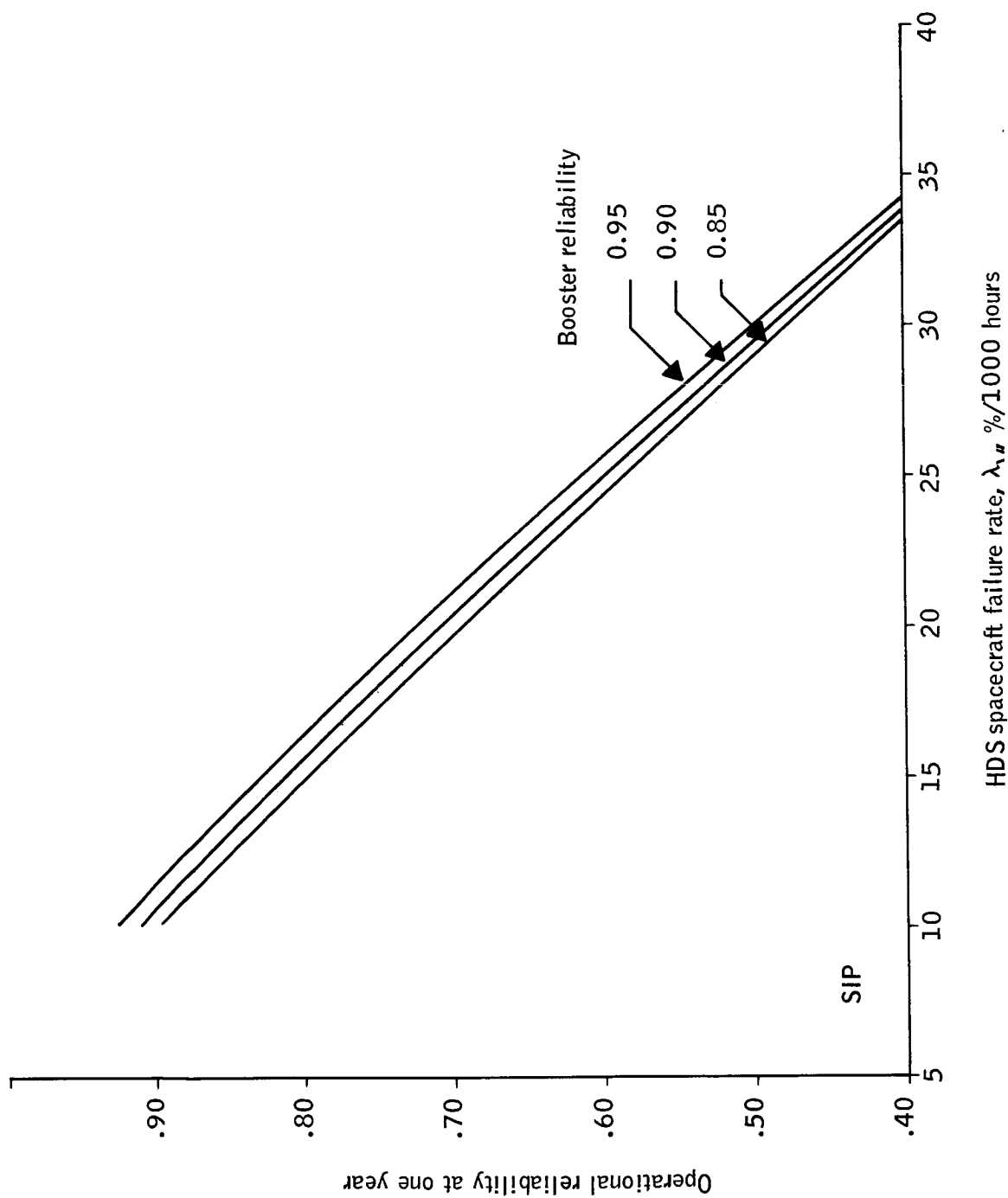


Figure B3. Mission Operational Reliability, Two Reserve Units Program Utilization

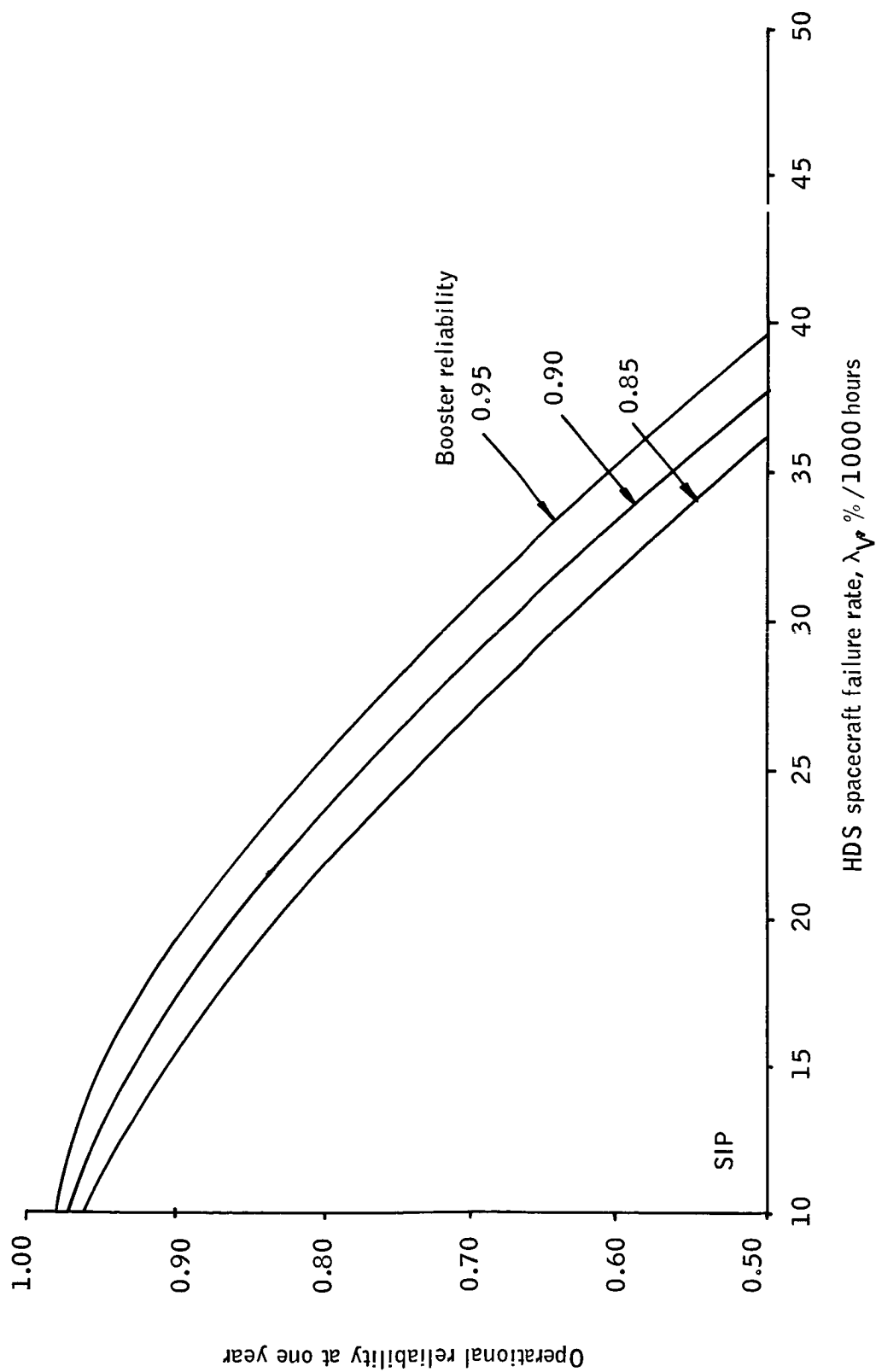


Figure B4. Mission Operational Reliability, Three Reserve Units Program Utilization

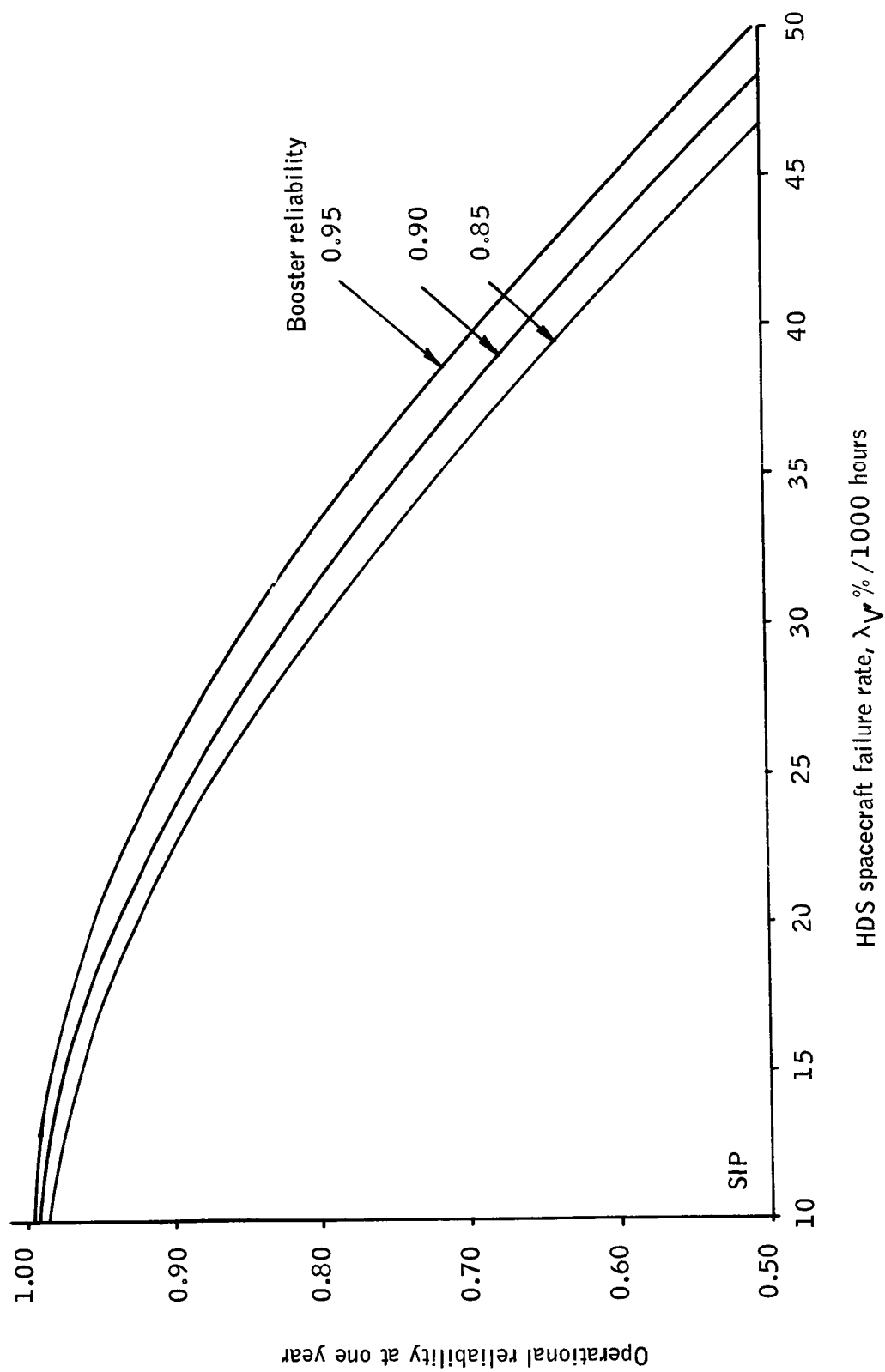


Figure B5. Mission Operational Reliability, Four Reserve Units Program Utilization

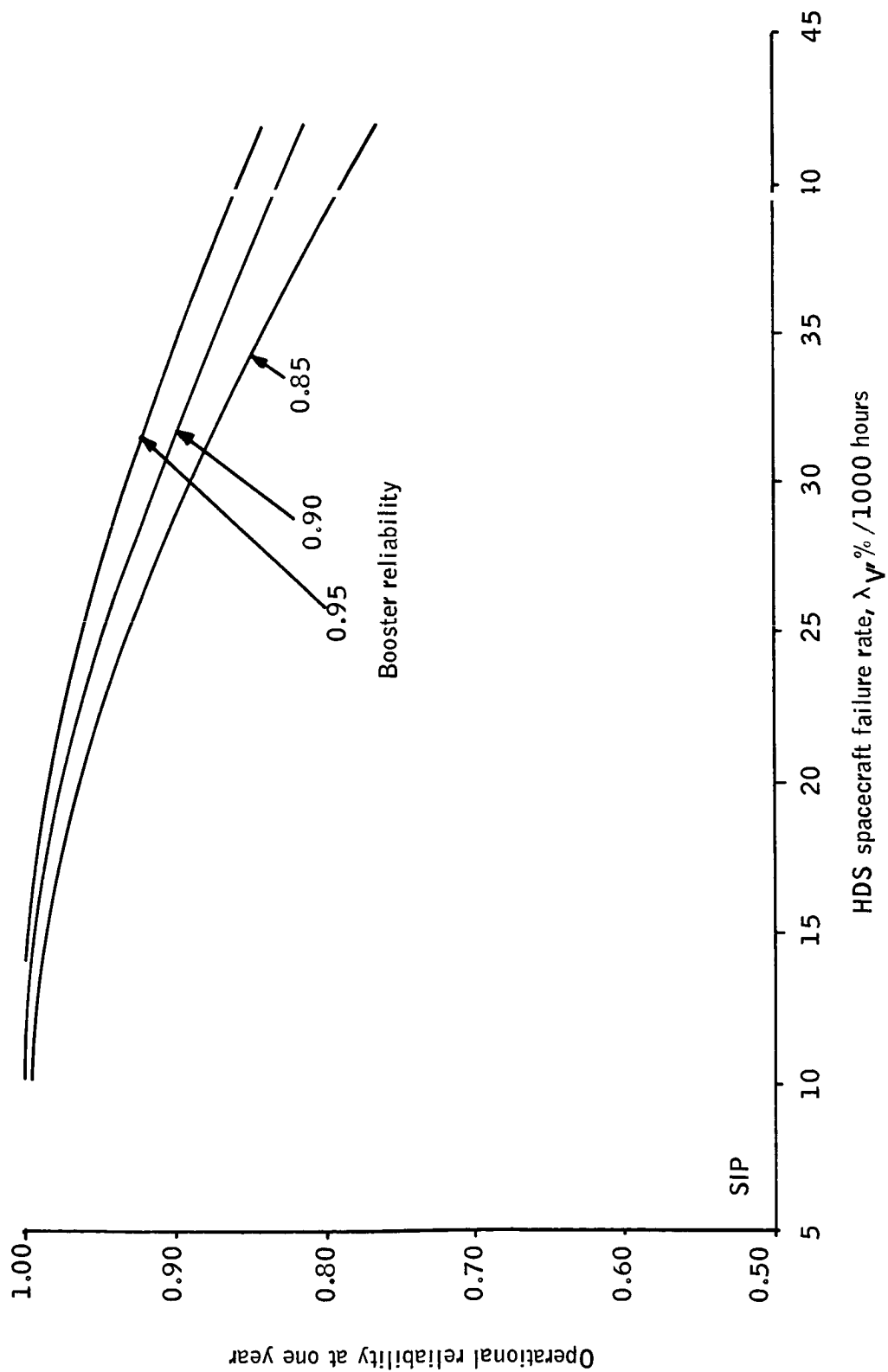


Figure B6. Mission Operational Reliability, Five Reserve Units Program Utilization

MISSION DATA FIGURE-OF-MERIT RELIABILITY

The following figures illustrate the variation in mission data figure-of-merit reliability with spacecraft failure rate for three assumed booster reliabilities and various numbers of reserve vehicles.

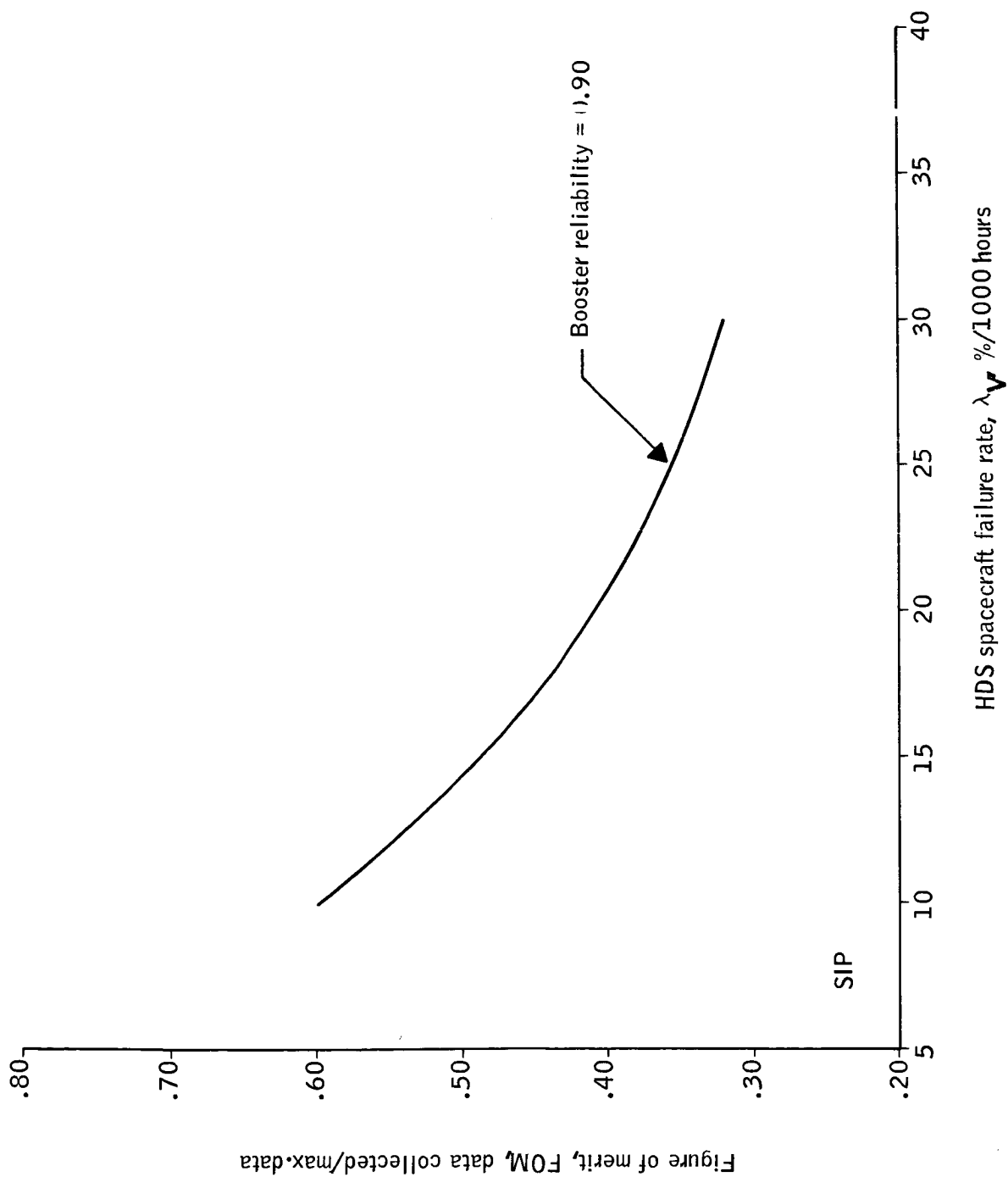
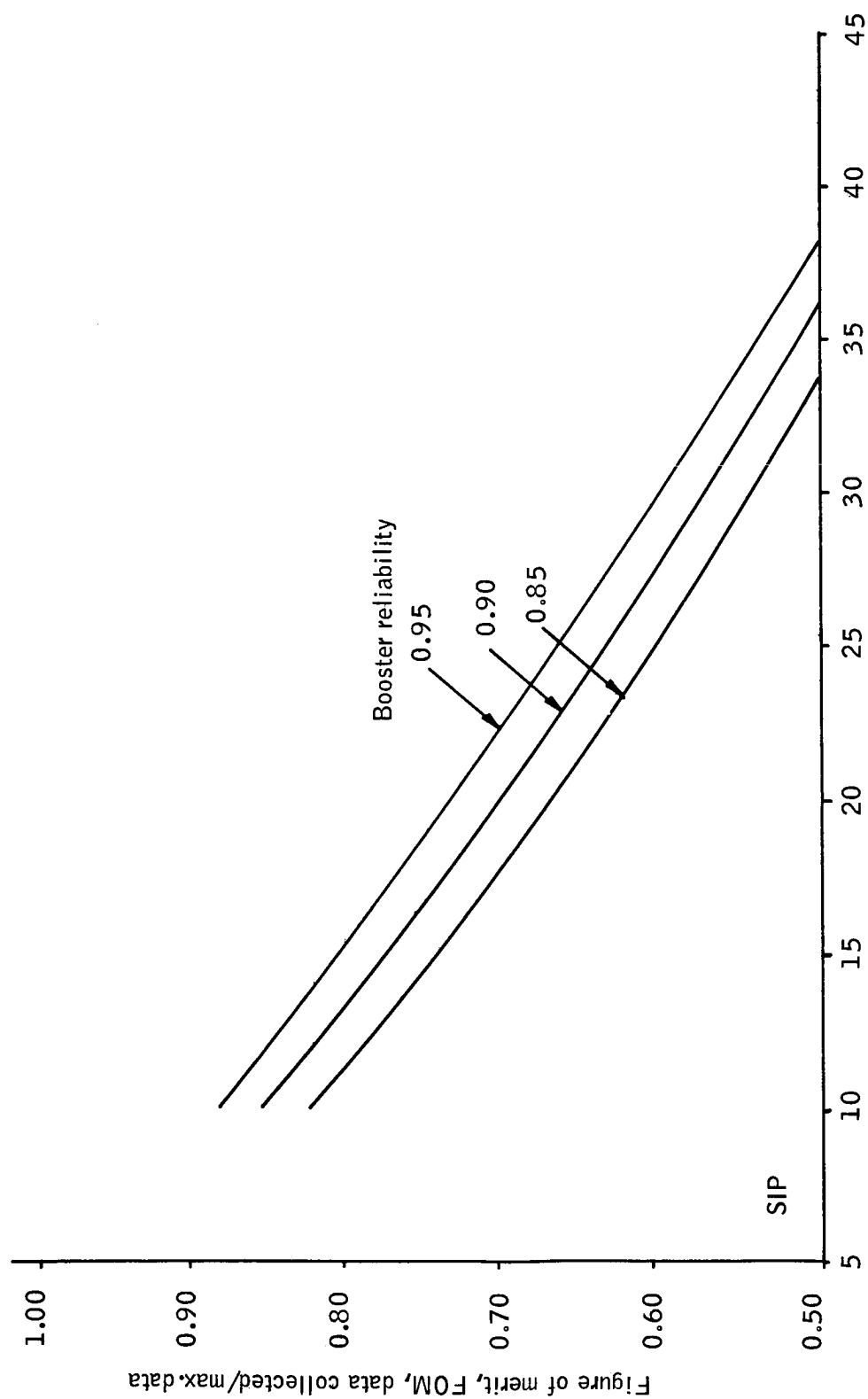


Figure B7. Mission Data Figure of Merit, No Reserve Program



HDS spacecraft failure rate, λ_{sp} %/1000 hours

Figure B8. Mission Data Figure of Merit, One Reserve Unit Program Utilization

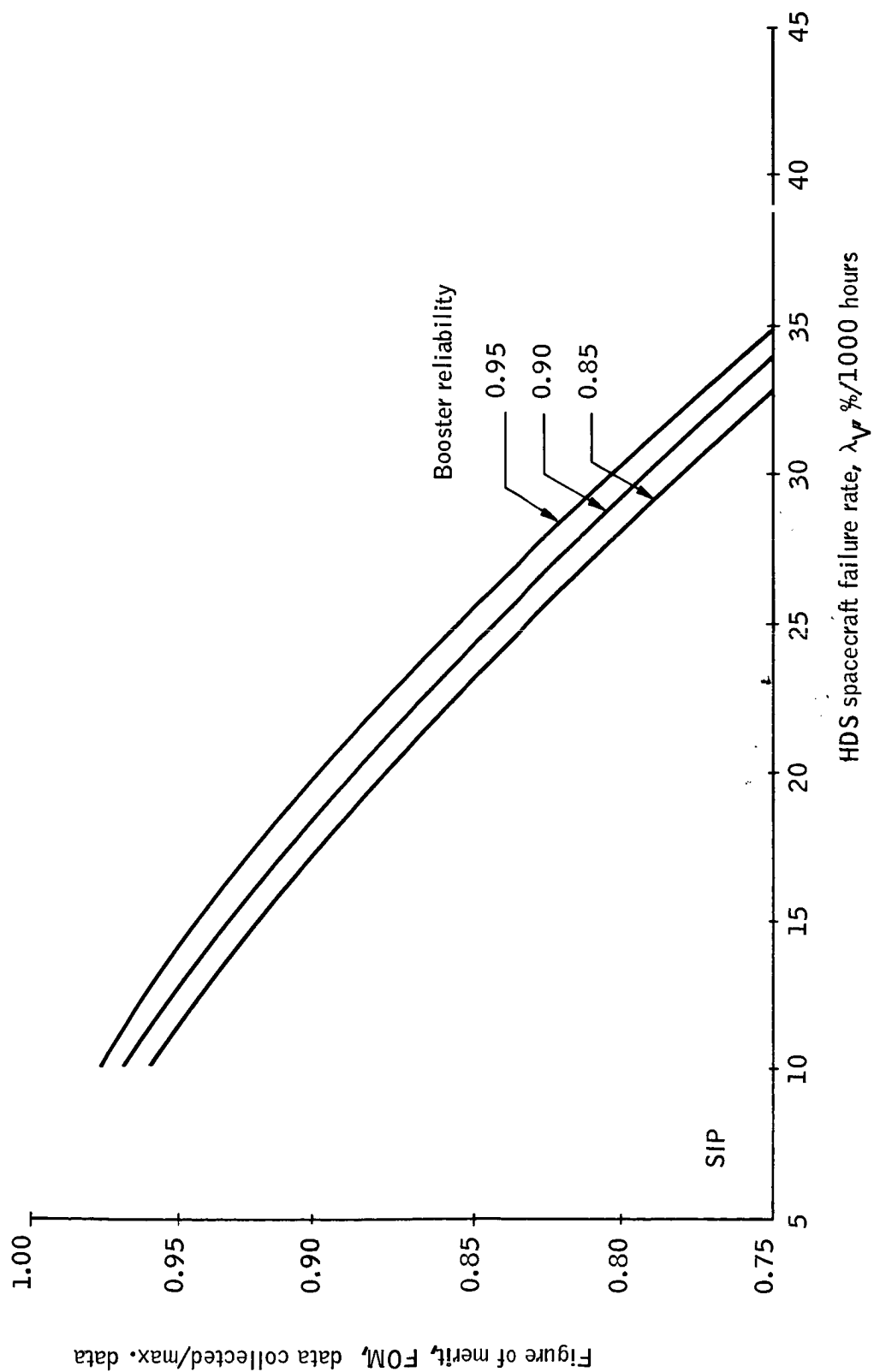


Figure B9. Mission Data Figure of Merit, Two Reserve Units Program Utilization

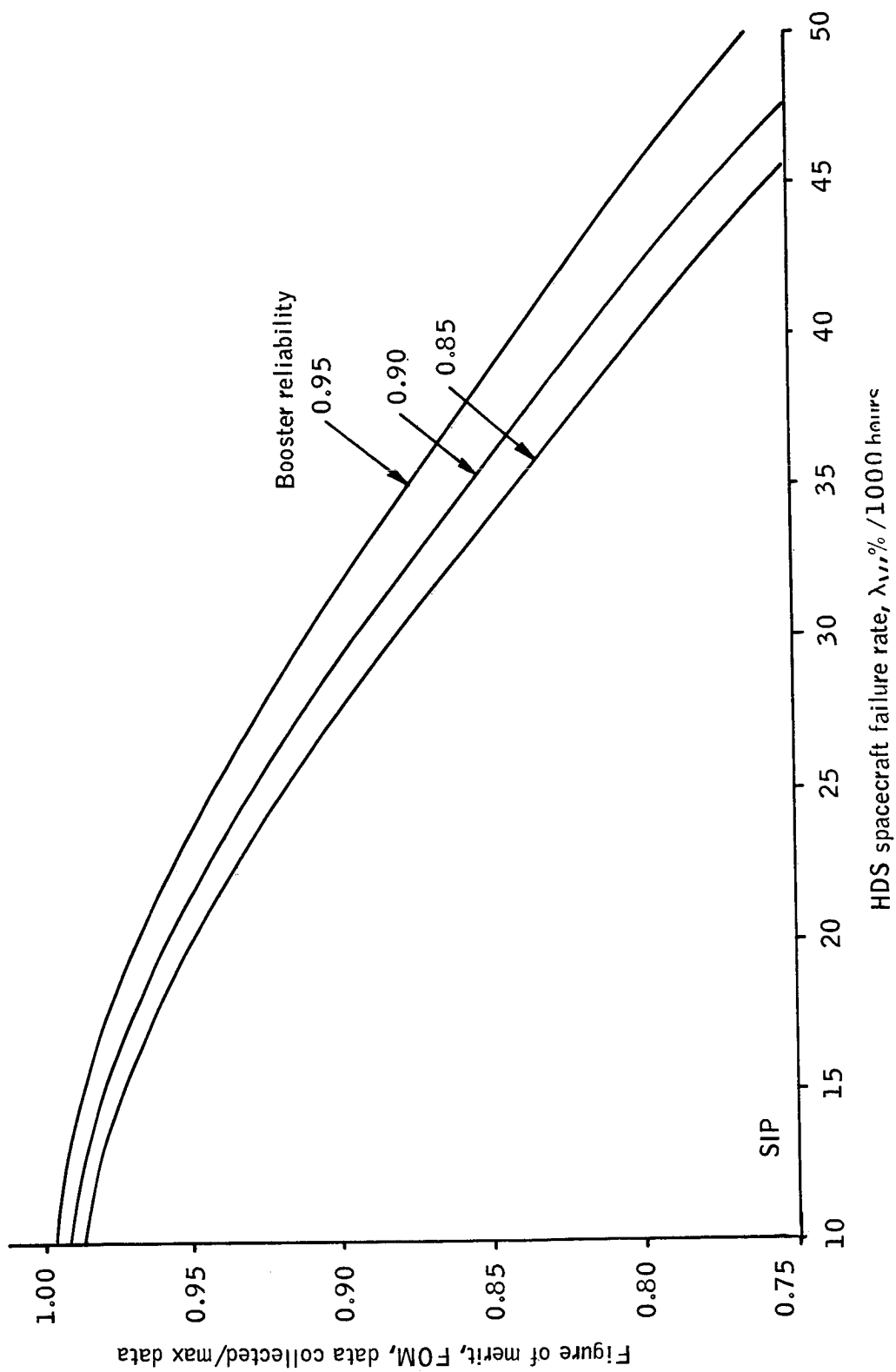


Figure B10. Mission Data Figure of Merit, Three Reserve Units Program Utilization

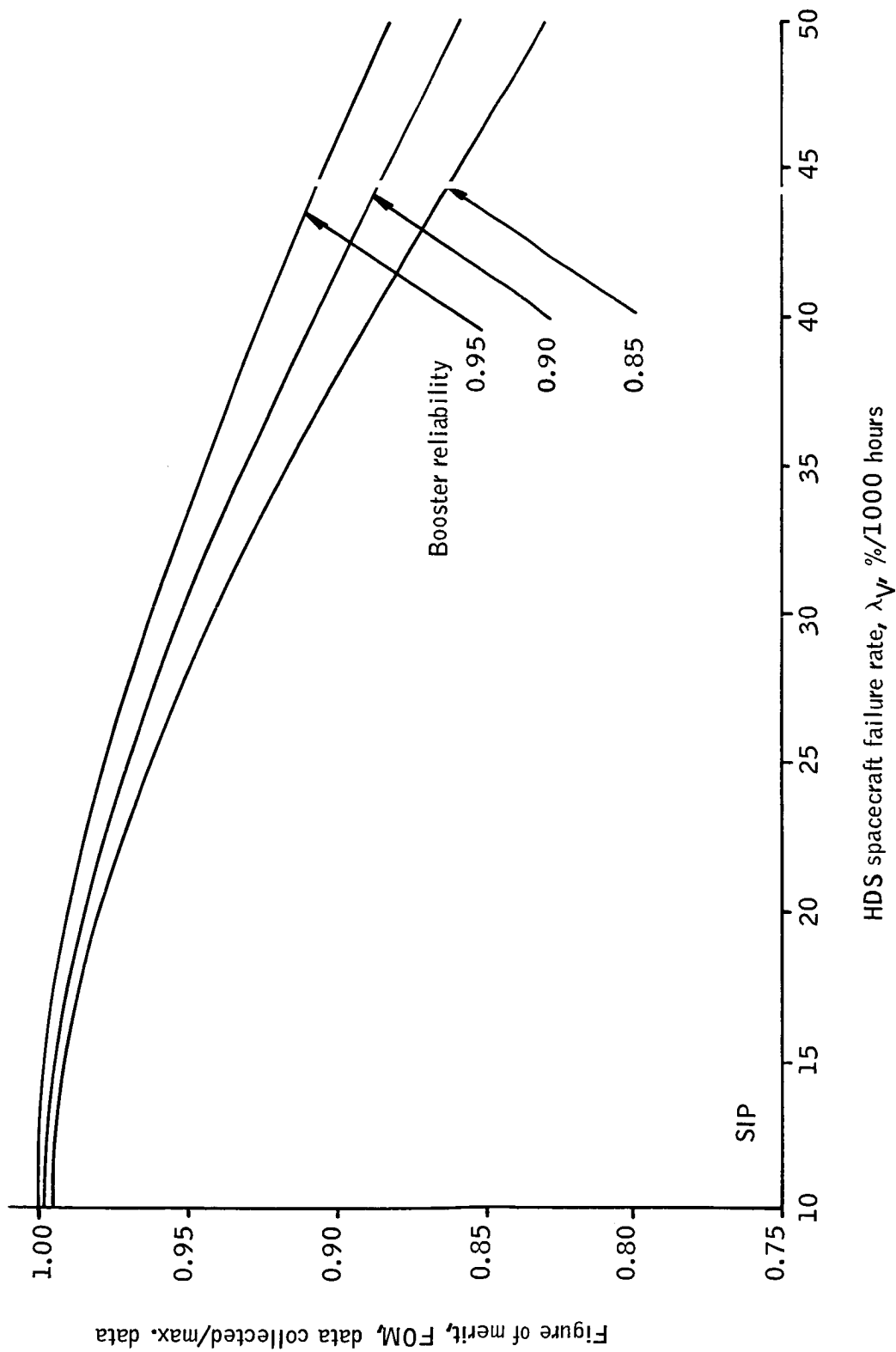


Figure B11. Mission Data Figure of Merit, Four Reserve Units Program Utilization

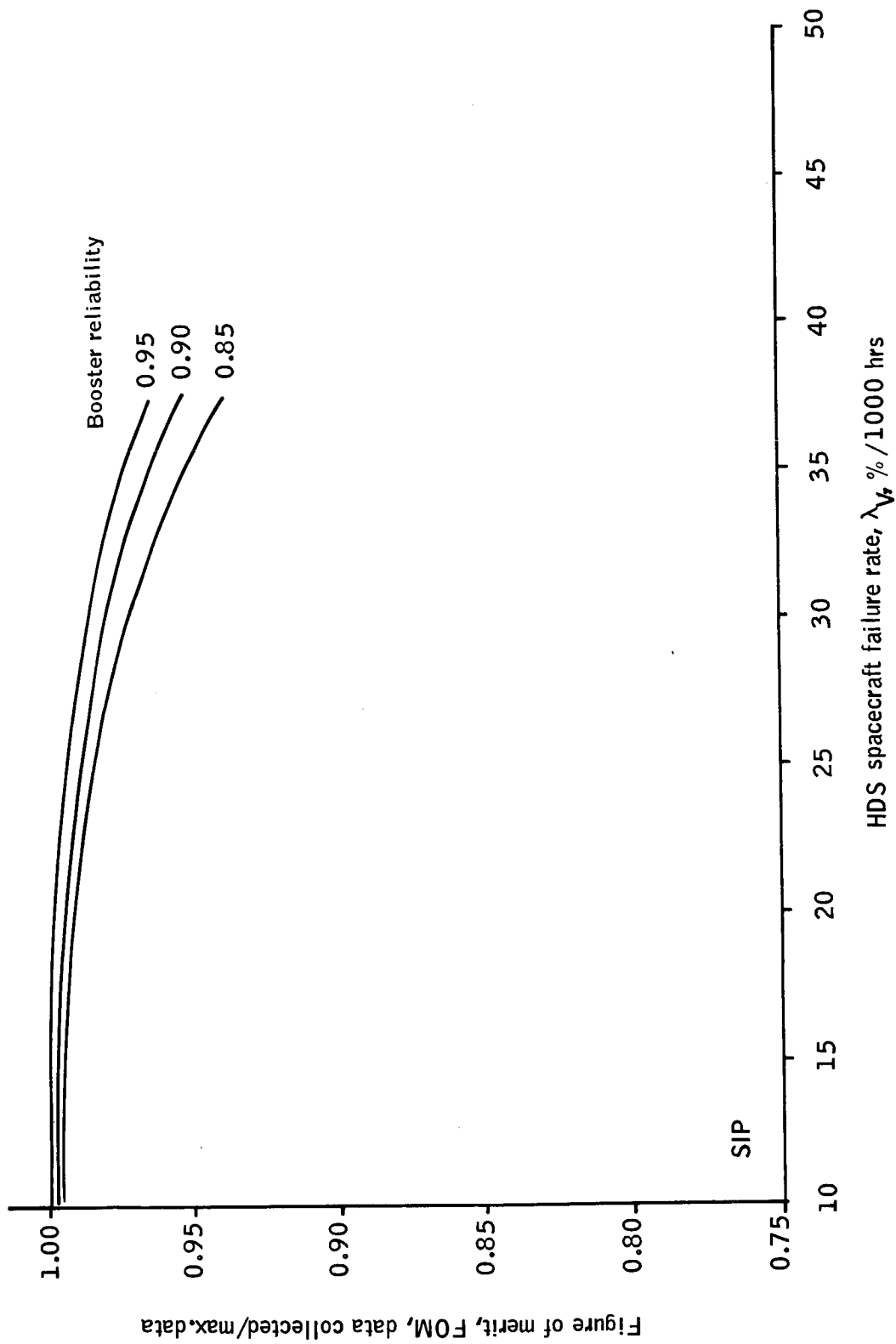


Figure B12. Mission Data Figure of Merit, Five Reserve Units Program Utilization

BOOSTER LAUNCH PROBABILITY SUMMARY

The following figures show the probability of the nth booster being used as a function of operating time in a program with a given number of boosters. Three spacecraft failure rates are shown on each figure and figures are presented for each of three booster reliabilities.

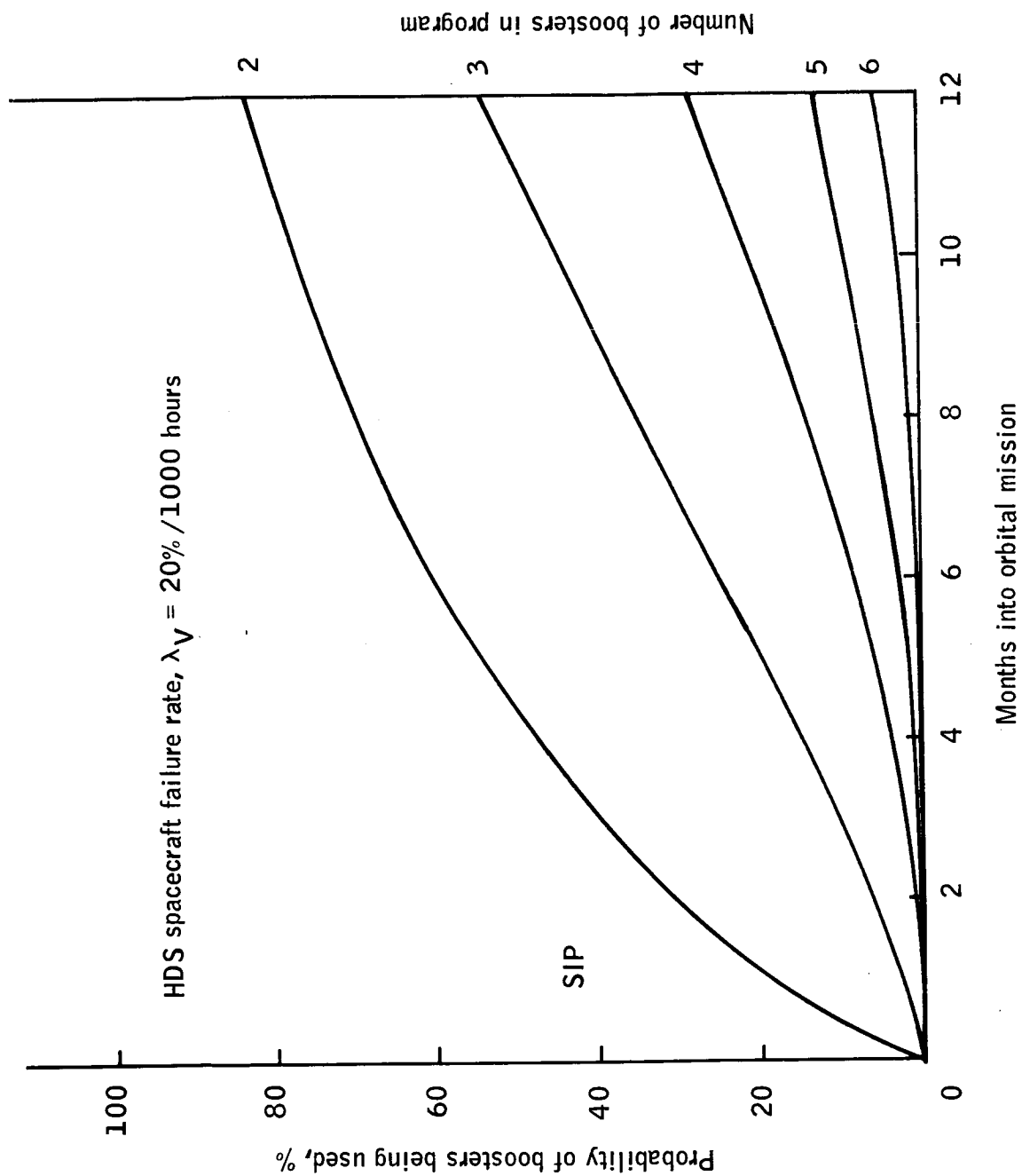


Figure B13. Booster Launch Probability, Booster Reliability = 0.95

BOOSTER LAUNCH PROBABILITIES

$$\underline{R_B = 0.85}$$

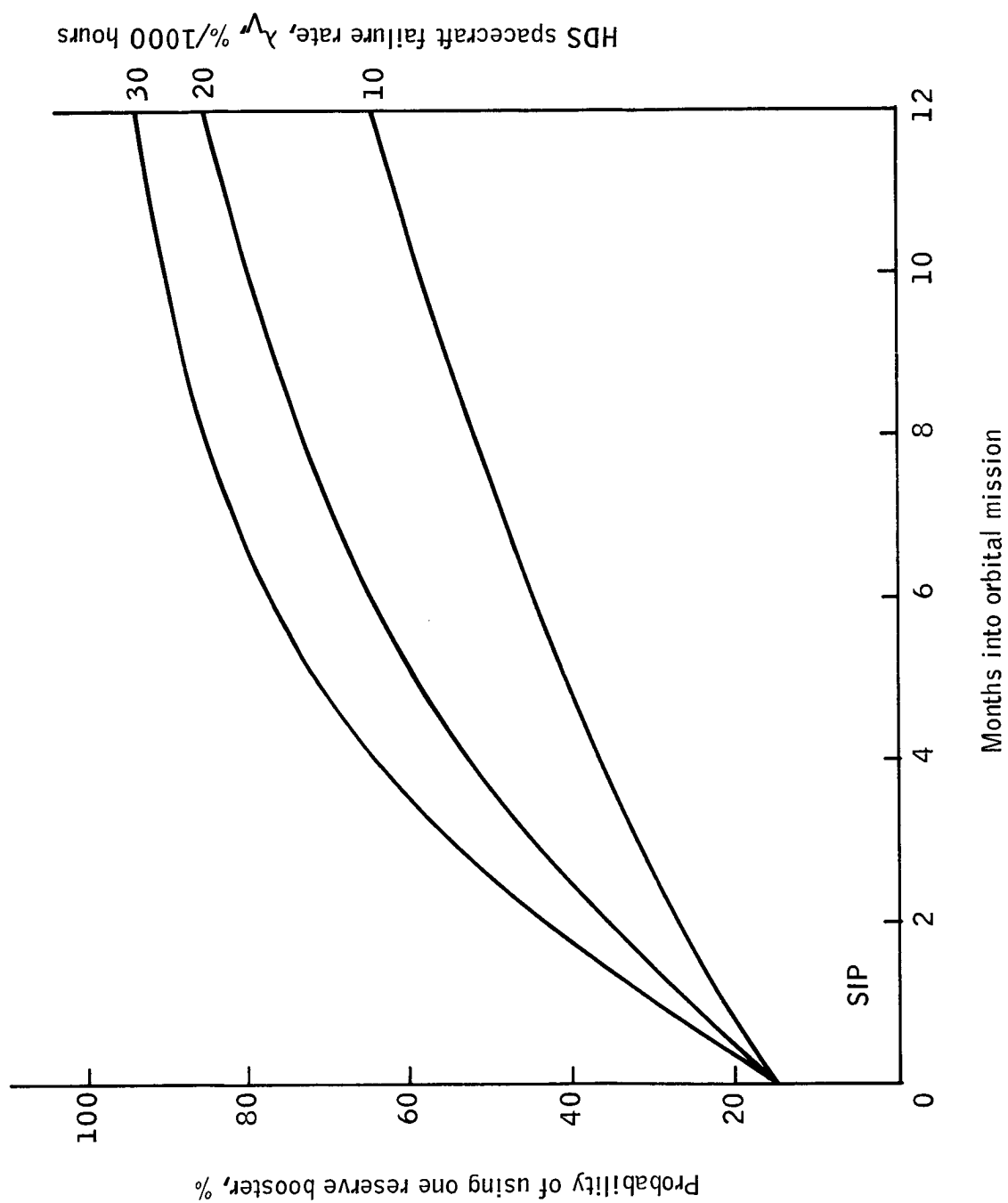


Figure B14. Booster Launch Probability, One Reserve, Booster Reliability = 0.85

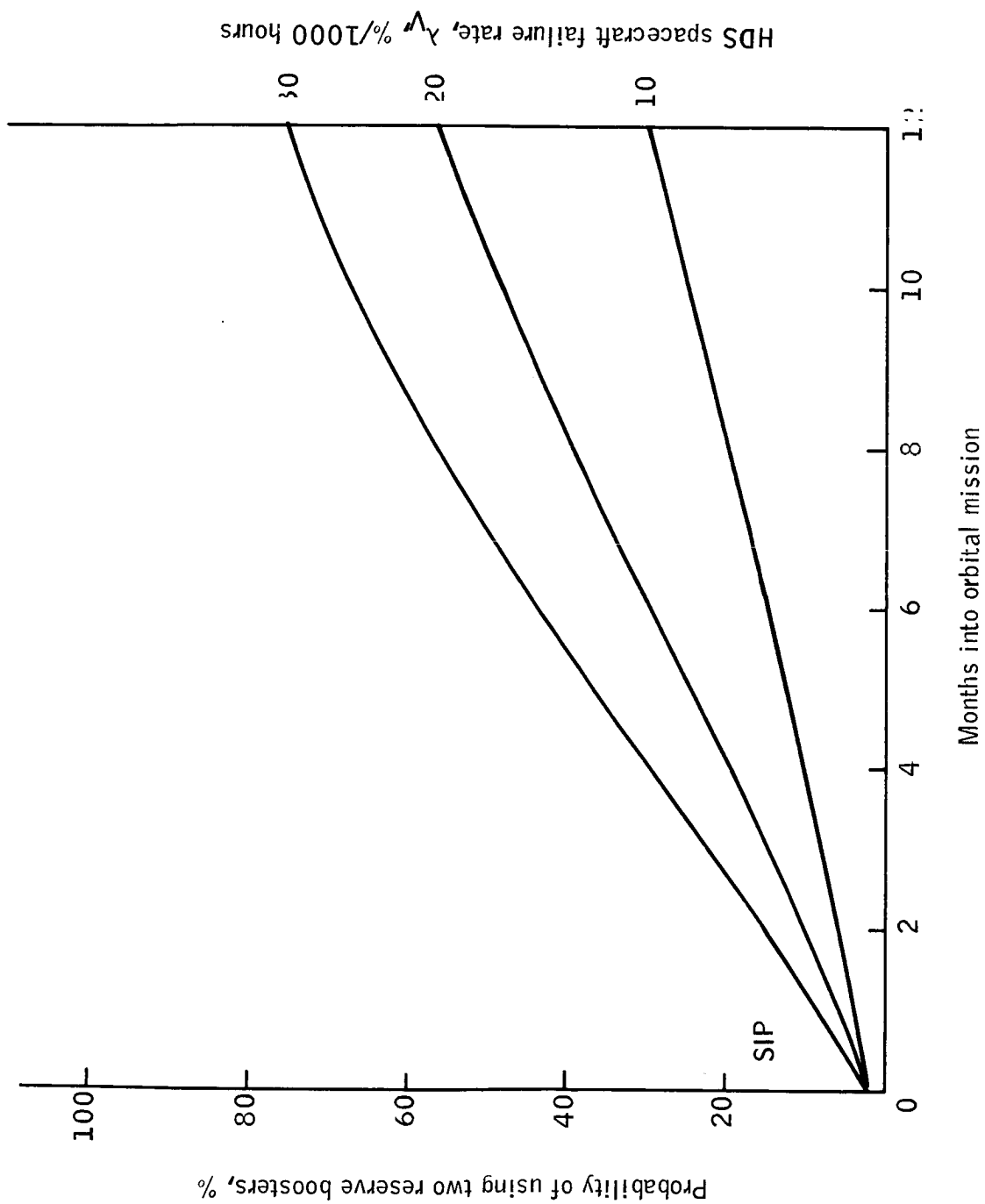


Figure B15. Booster Launch Probability, Two Reserves, Booster Reliability = 0.85

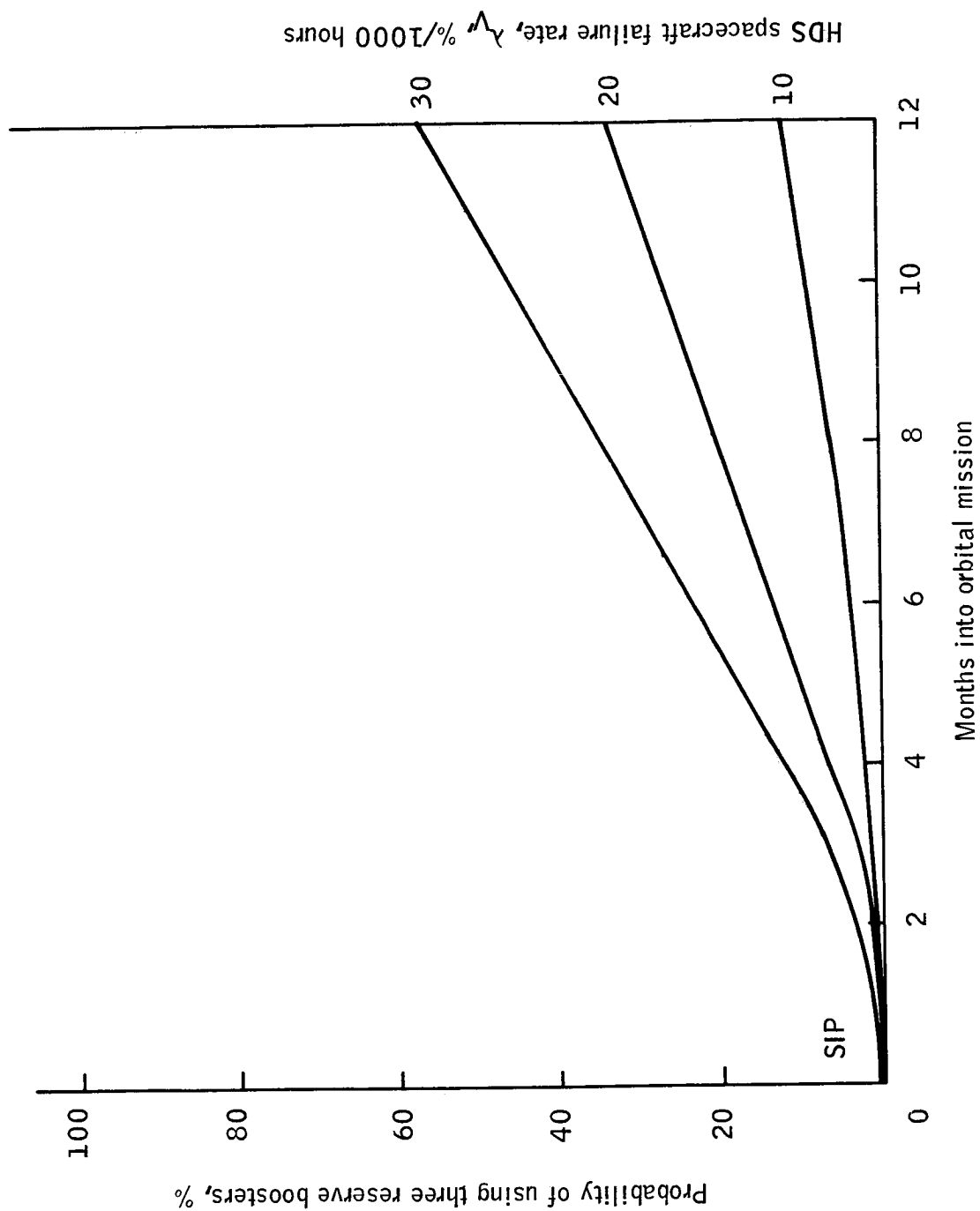


Figure B16. Booster Launch Probability, Three Reserves, Booster Reliability = 0.85

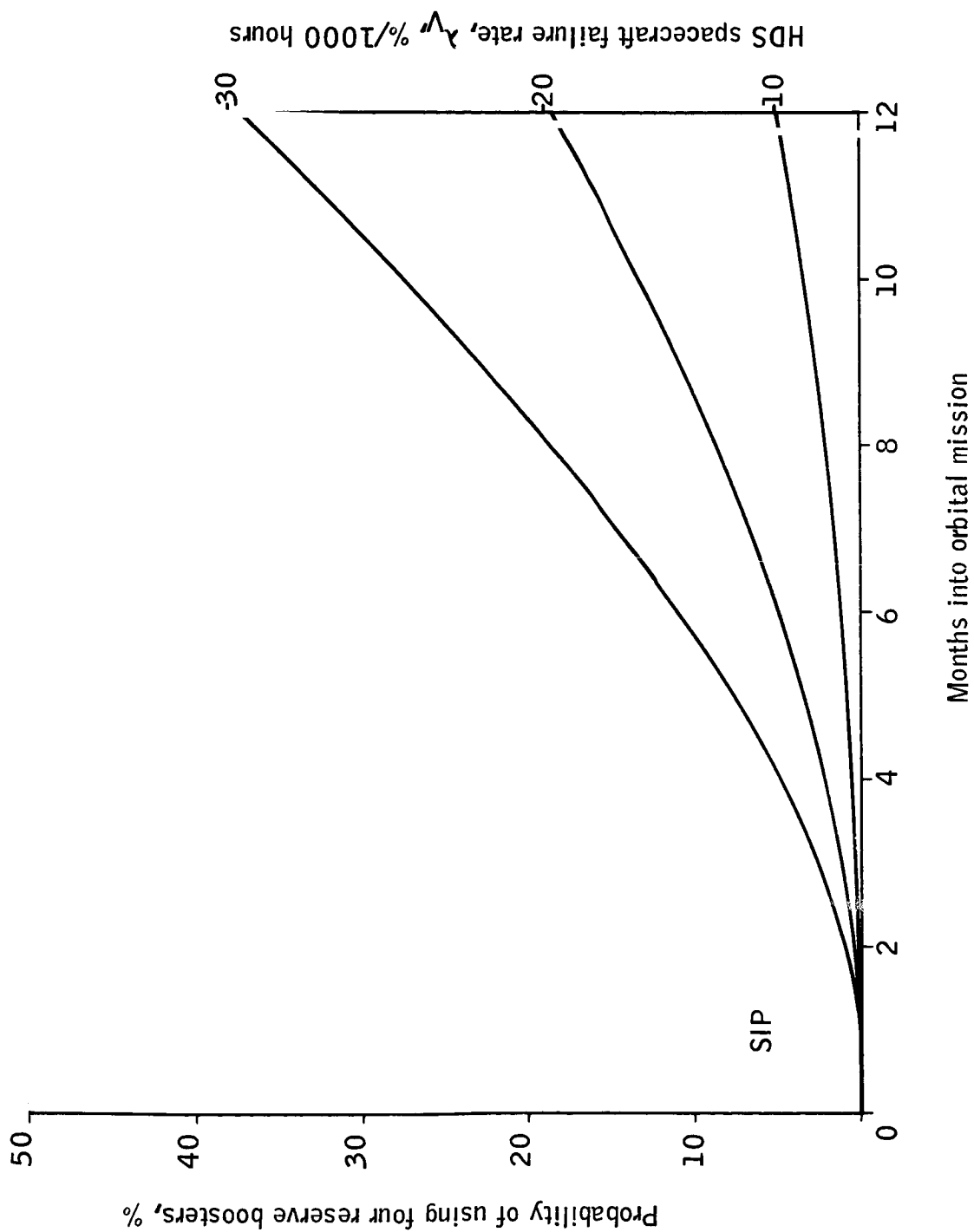


Figure B17. Booster Launch Probability, Four Reserves, Booster Reliability = 0.85

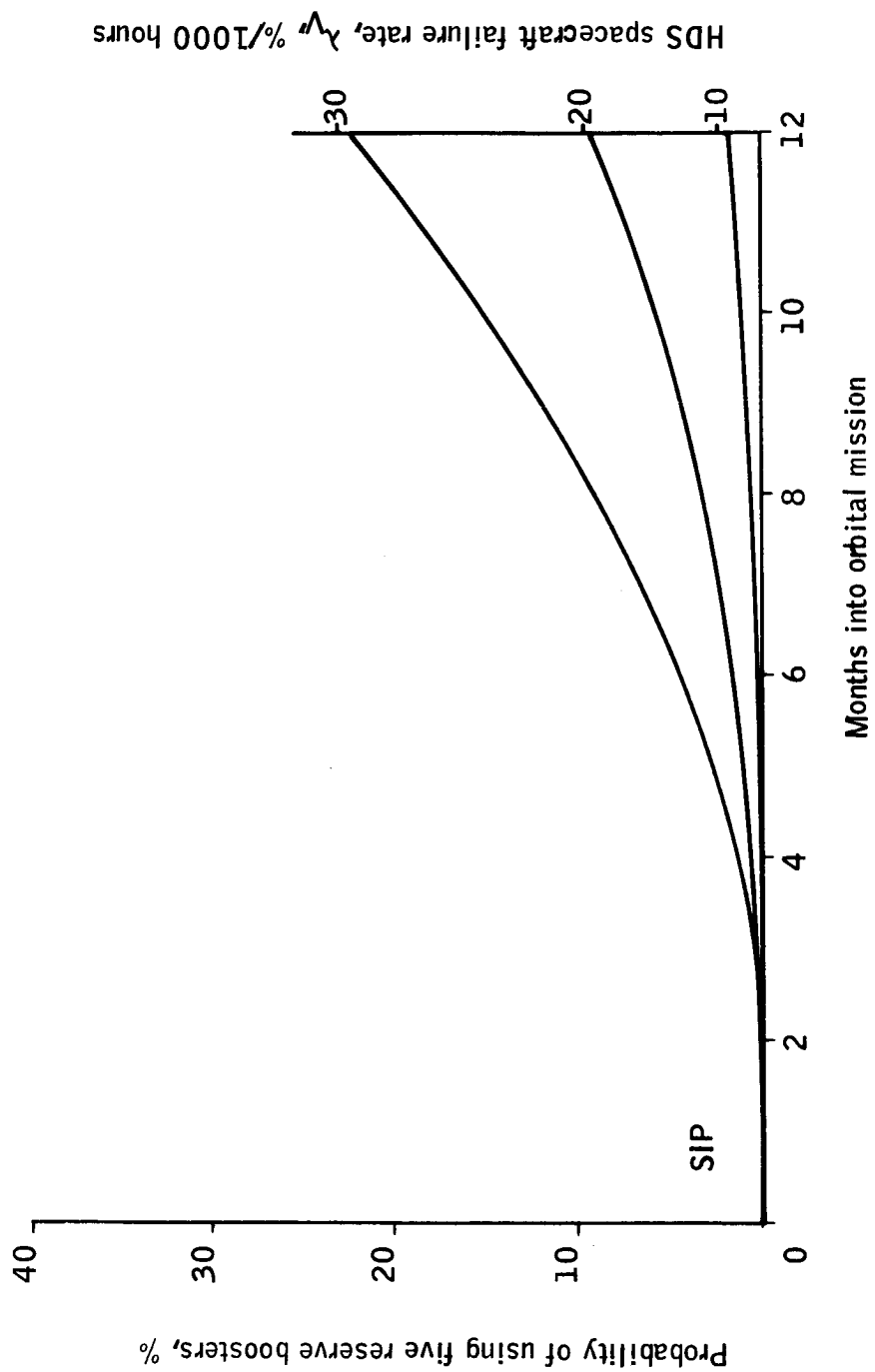


Figure B18. Booster Launch Probability, Five Reserves, Booster Reliability = 0.85

BOOSTER LAUNCH PROBABILITIES
 $R_B = 0.90$

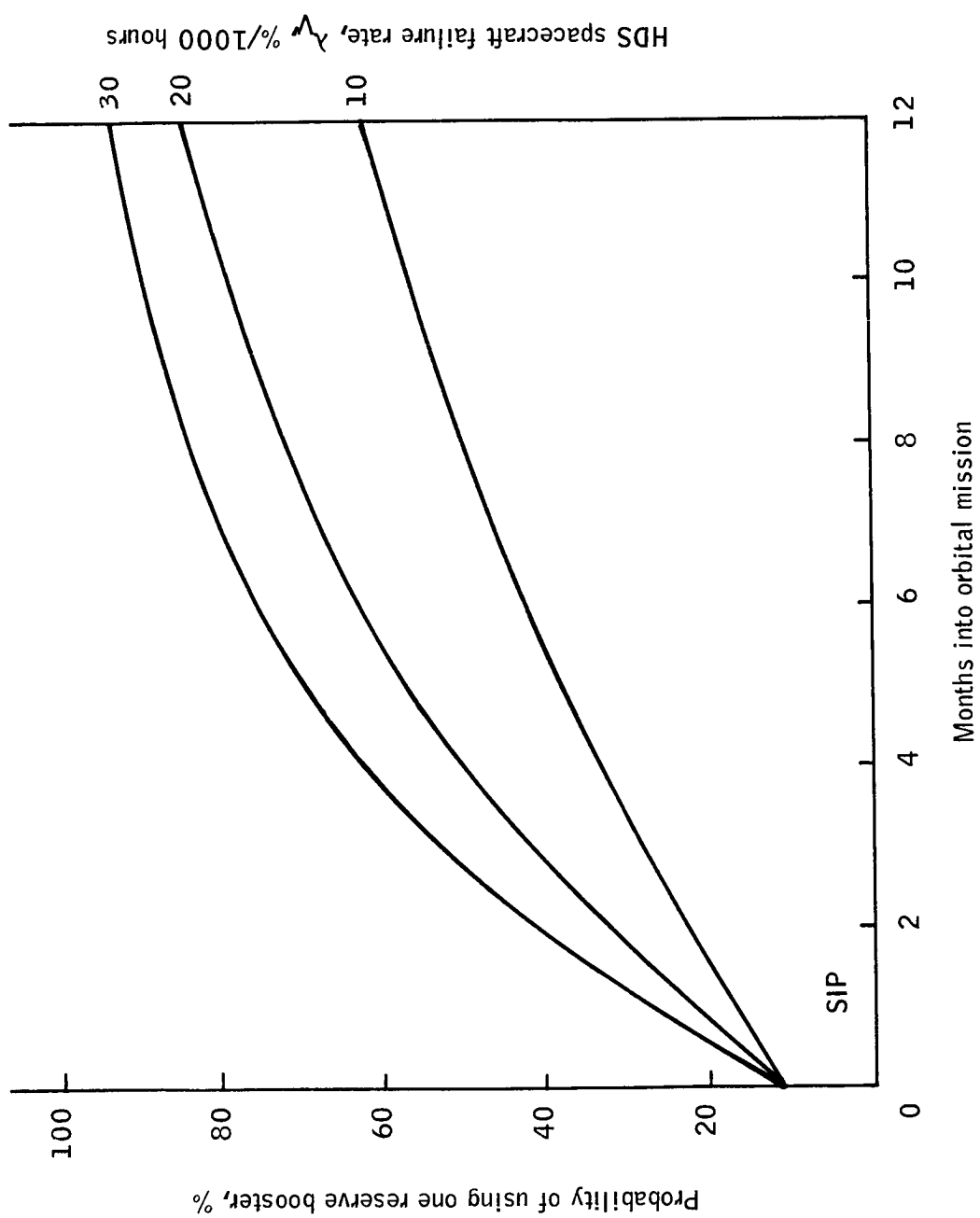


Figure B19. Booster Launch Probability, One Reserve, Booster Reliability = 0.9C

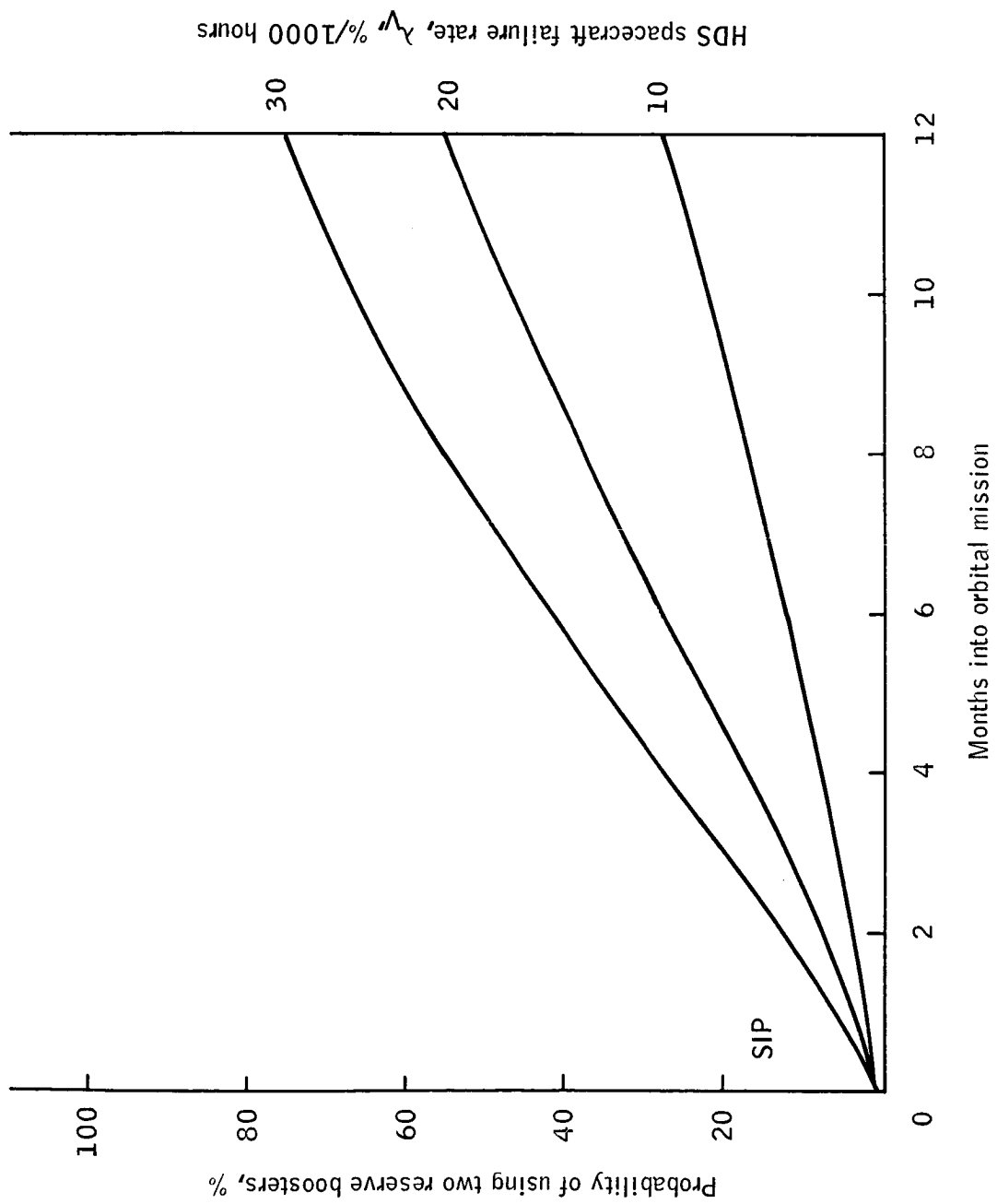


Figure 320. Booster Launch Probability, Two Reserves, Booster Reliability = 0.99

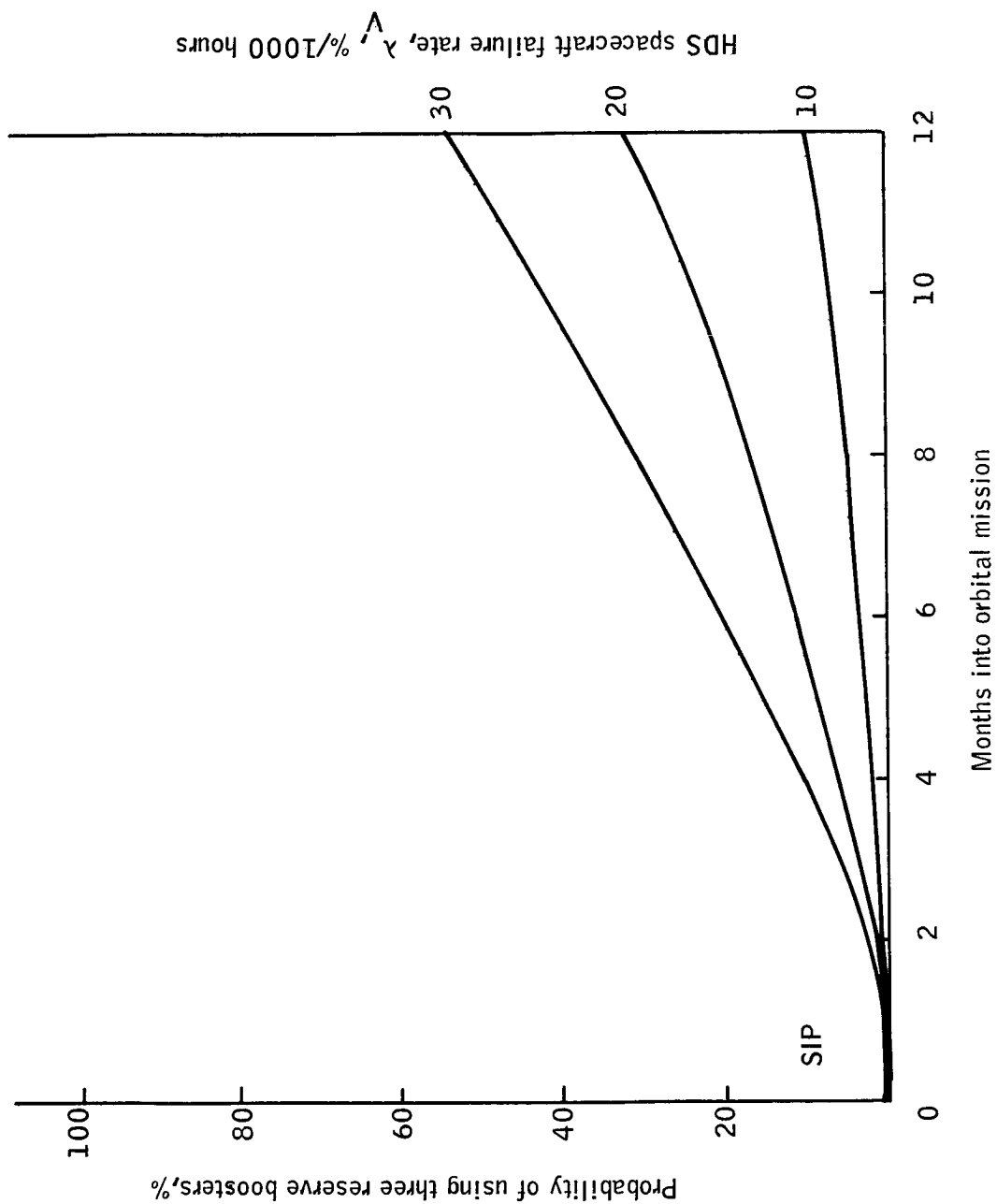


Figure B21. Booster Launch Probability, Three Reserves, Booster Reliability = 0.90

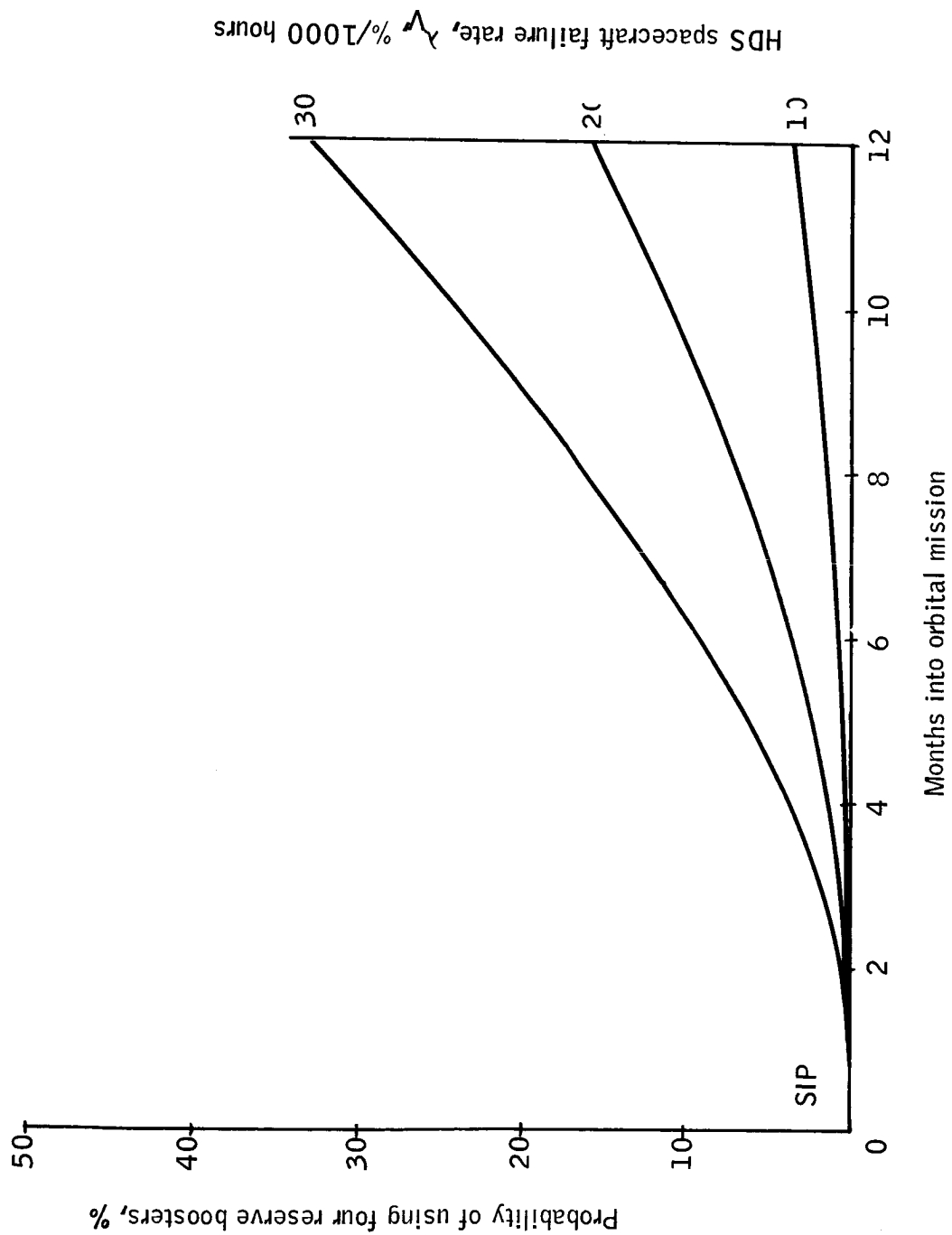


Figure B22. Booster Launch Probability, Four Reserves, Booster Reliability = 0.90

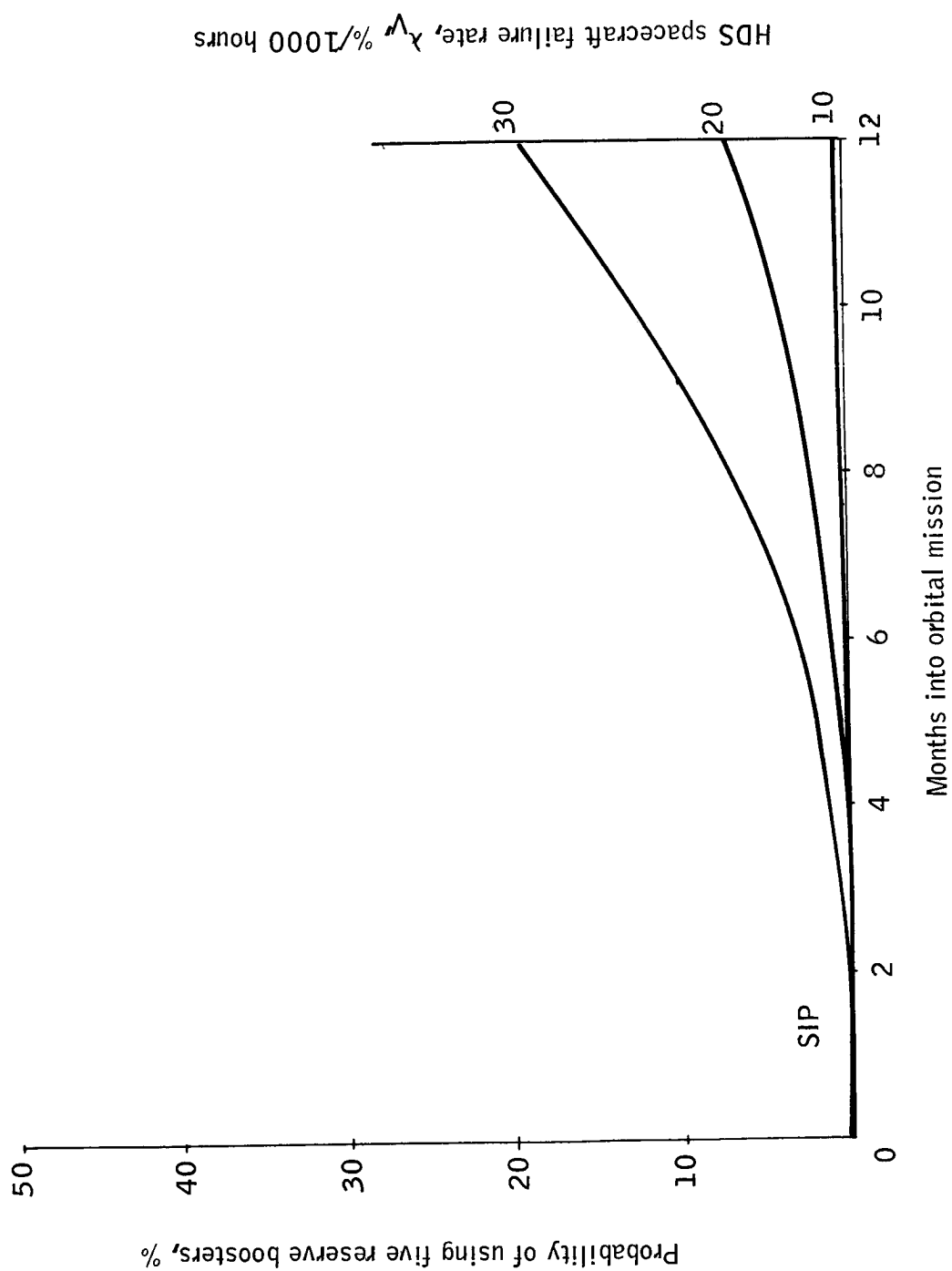


Figure B23. Booster Launch Probability, Five Reserves,
Booster Reliability = 0.90

BOOSTER LAUNCH PROBABILITIES

$$\underline{R_B = 0.95}$$

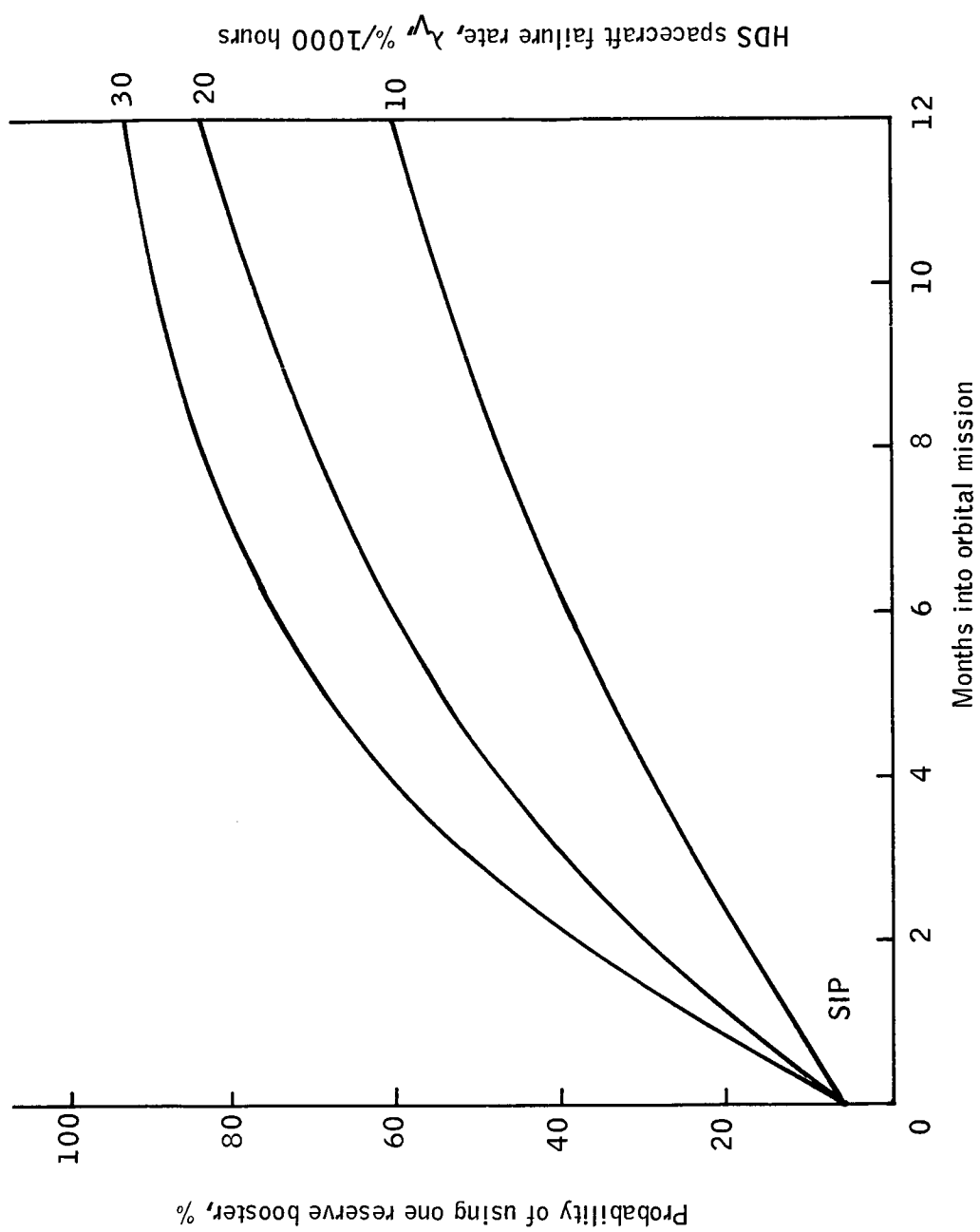


Figure B24. Booster Launch Probability, One Reserve,
Booster Reliability = 0.95

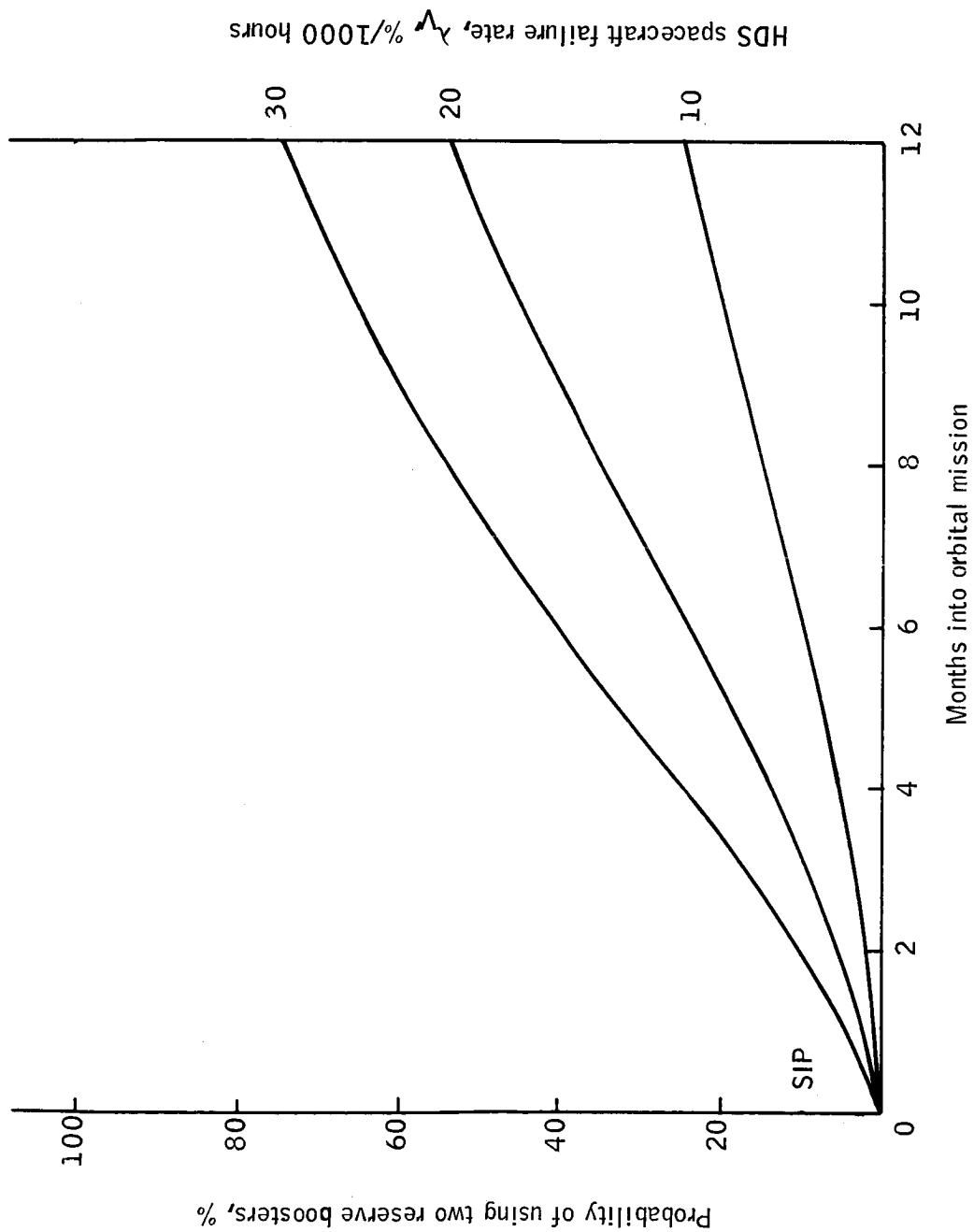


Figure B25. Booster Launch Probability, Two Reserves,
Booster Reliability = 0.95

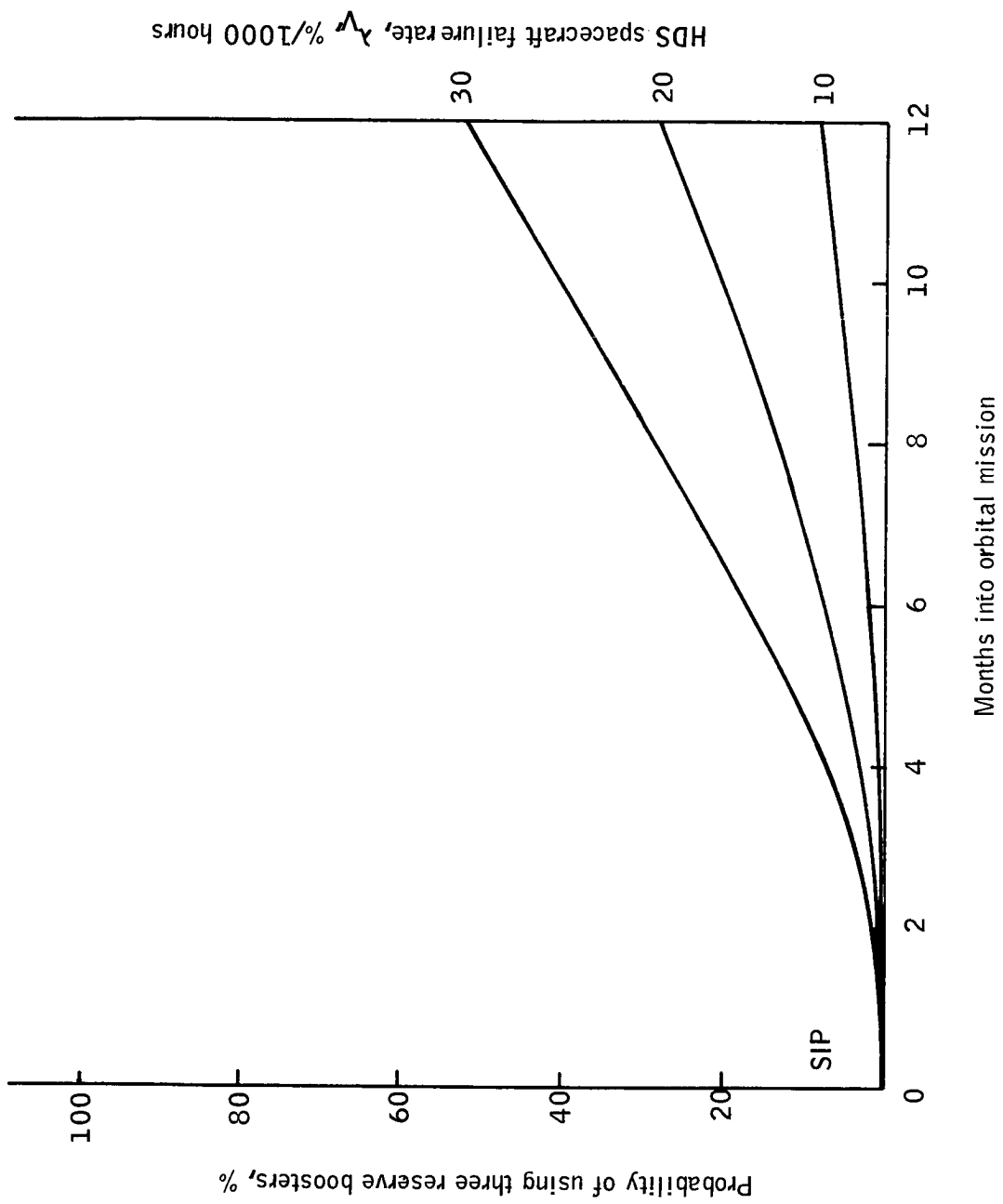


Figure B26. Booster Launch Probability, Three Reserves,
Booster Reliability = 0.95

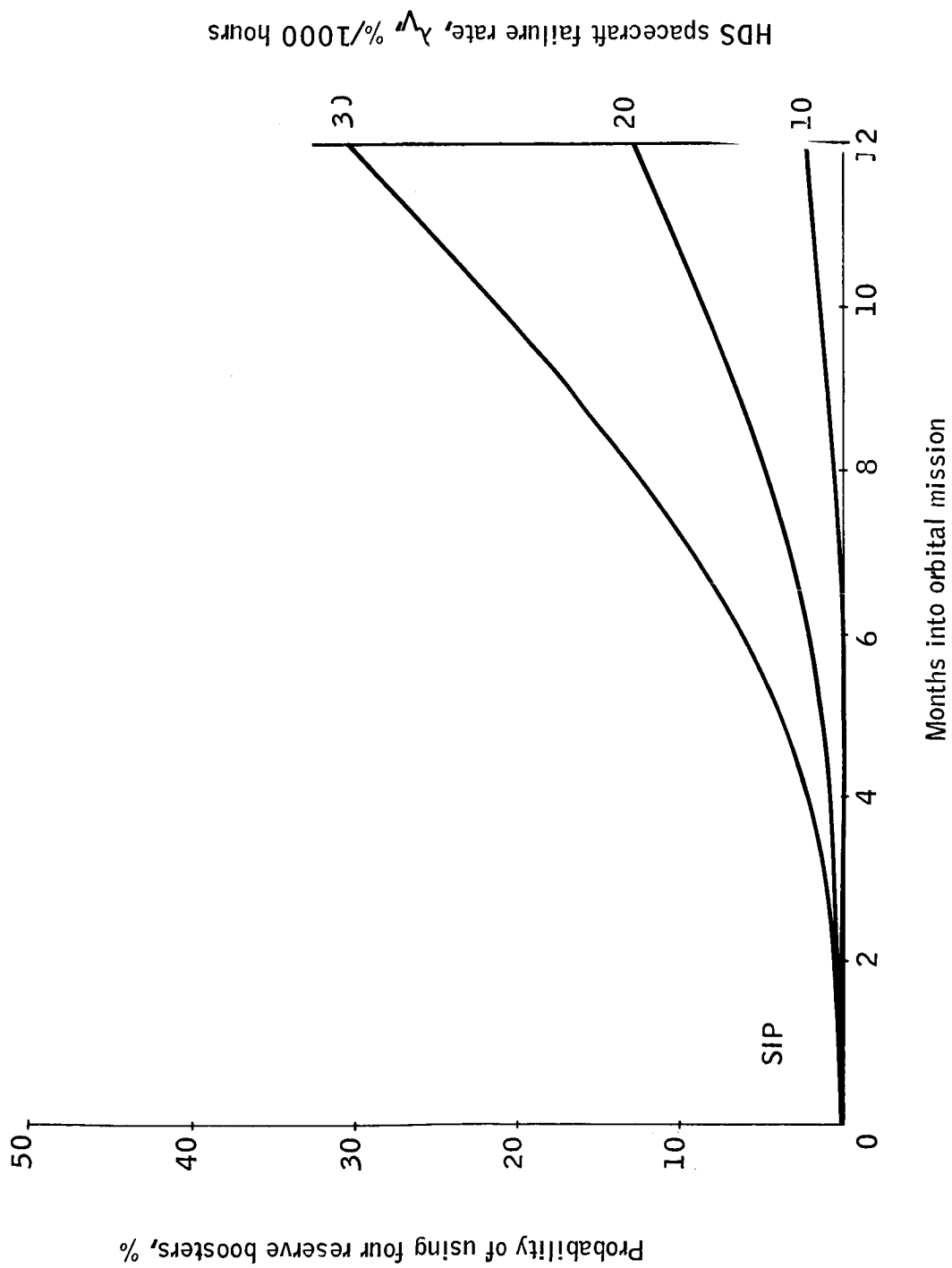


Figure B27. Booster Launch Probability, Four Reserves,
Booster Reliability = 0.95

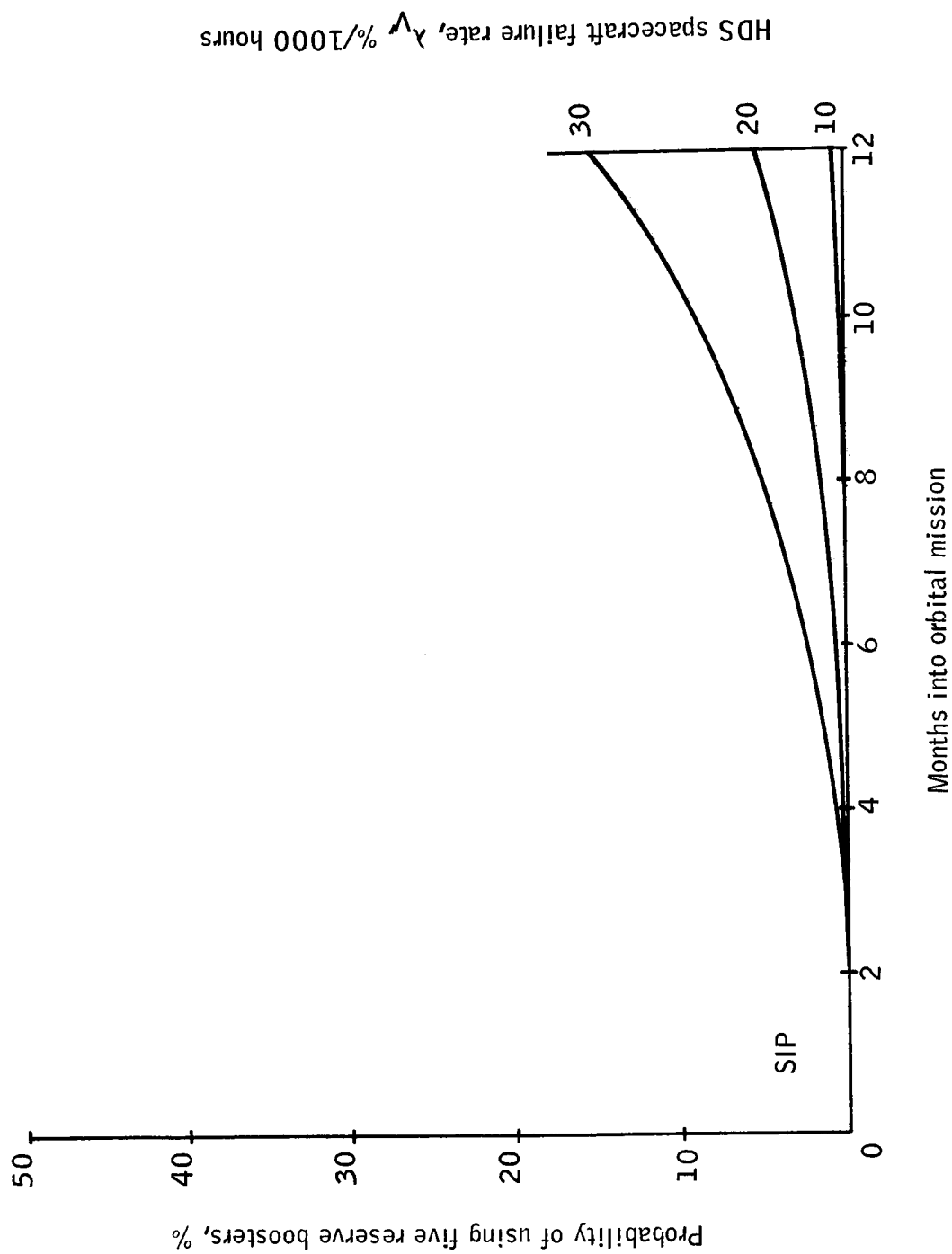


Figure B28. Booster Launch Probability, Five Reserves,
Booster Reliability = 0.95